APOLLO RENDEZVOUS SIMULATOR STUDY

CONTRACT NASw -413(HS-625)

TECHNICAL DATA REPORT

VOLUME III

VEHICLE ANALYSIS - PROPULSION

12 JULY 1962

REPORT NO. 324.4

ASTRONAUTICS DIVISION
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VEHICLE ANALYSIS - PROPULSION

Apollo Rendezvous Simulator Study

Contract NASw - 413(HS-625)

Submitted By:

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to

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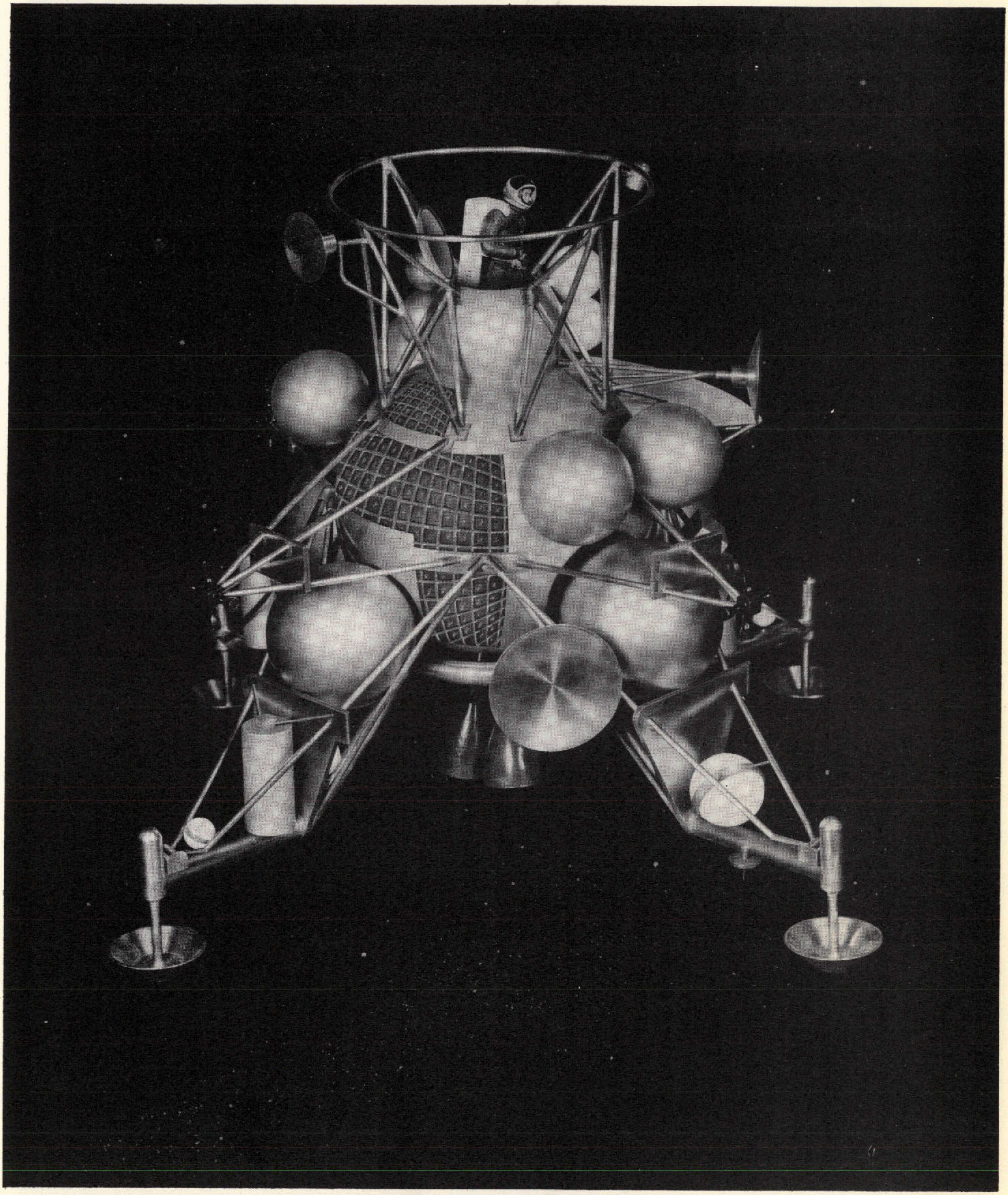
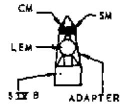
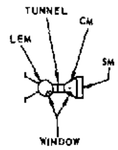
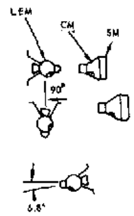
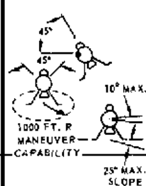
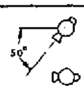
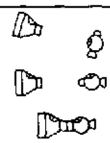


LUNAR EXCURSION MODULE

MISSION PHASE	PHASE IS TERMINATED BY	EVENT SEQUENCE	LEM ORIENTATION	TIME EVENT	TIME CUM (hrs)	ALTITUDE	DELTA V (Ideal) fps	LEM WEIGHT (Earth lbs)
EARTH LAUNCH & PARKING ORBIT	BURNOUT AT TRANSLUNAR INJECTION	● BOOST INTO EARTH PARKING ORBIT		300 sec.	.1	0	29,760	29,290
		● COAST IN PARKING ORBIT		1.5 hrs.	1.6	150 NM (earth)		
		● TRANSLUNAR INJECTION FROM EARTH PARKING ORBIT		135 sec	1.7		10,700	
TRANSLUNAR	START OF RETRO INTO LUNAR ORBIT	● COAST						
		● SEPARATE & JETTISON ADAPTOR						
		● DOCK COMMAND MODULE – SERVICE MODULE INTO LEM						
		● SEPARATE SIV B FROM LEM						
		● PERFORM MIDCOURSE CORRECTION (S)					400	
		● COAST TO START OF LUNAR ORBIT INJECTION		72 hrs.	73.7			29,290
LUNAR ORBIT & ORBIT TRANSFER	START OF GROSS DECELERATION	● RETRO INTO LUNAR ORBIT		200 sec.	73.8	59 NM (Lunar)	3,000	
		● ESTABLISH ORBIT EPHEMERIS						
		● TRANSFER ONE CREW MEMBER FROM COMMAND MODULE & INITIATE CHECKOUT OF LEM						
		● TRANSFER OTHER CREW MEMBER & COMPLETE CHECKOUT						
		● COAST TO RETRO POINT, OVERPASS LANDING SITE ONCE						
		● SEPARATE LEM FROM CM		150 min.	76.3			29,894
		● RETRO TO INITIATE TRANSFER ORBIT		26 sec.	76.3	59 NM	255	
		● COAST IN TRANSFER ORBIT						
		● OBTAIN NAVIGATION DATA		29 min.	76.8			
LUNAR DESCENT	LUNAR TOUCHDOWN	● INITIATE GROSS DECELERATION		333 sec.	76.9	62,000 FT	5,912	
		● HOVER & LATERAL MOVEMENT. OBSERVE LANDING SITE				100 FT	590	
		● DESCENT TO SURFACE		105 sec.	77.0	10 FT		15,157
LUNAR SURFACE STAY	START OF LUNAR LAUNCH	● STAGE LANDING GEAR ASSEMBLY						
		● INSPECTION OF LEM STRUCTURES & SYSTEMS						
		● EXPLORATION & EXPERIMENTATION						
		● MAINTENANCE AS REQUIRED						
		● CHECKOUT & COUNTDOWN		24 hrs.	101.0			11,042
LUNAR ASCENT & ORBIT TRANSFER	COMPLETION OF PLANE CHANGE WITH MAIN ENGINE	● GROSS LAUNCH AND INJECTION HOHMANN TRANSFER ORBIT		228 sec.	101.1	50,000 FT.	5,780	
		● COAST TOWARD ORBIT CONTAINING CM–SM & EXECUTE PLANE CHANGE WITH MAIN ENGINE		43.7 min.	101.8		25	
LUNAR ORBIT RENDEZVOUS	START OF TRANSEARTH INJECTION	● COAST TO ORBIT CONTAINING CM–SM				70,000 FT	193	
		● TERMINAL RENDEZVOUS MANEUVERS		13.3 min.	102.0			
		● DOCKING		2.7 min.	102.1	59 NM	10	5,972
		● CREW TRANSFER FROM LEM TO CM		14.5 min.	102.3			
		● SEPARATE & ABANDON LEM – COAST TO TRANSEARTH INJECTION POINT		42.8 min.	103.0			5,248
TRANSEARTH	RE-ENTRY INTO EARTH'S ATMOSPHERE	● TRANSEARTH INJECTION OF COMMAND & SERVICE MODULES	NOT APPLICABLE	200 sec.	103.1		3,000	N.A.
		● COAST						
		● EFFECT MIDCOURSE CORRECTION(S)					400	
		● SEPARATE & JETTISON SERVICE MODULE						
		● COAST PRIOR TO RE-ENTRY		72 hrs.	175.1			
EARTH RE-ENTRY & LANDING	EARTH LANDING RECOVERY	● RE-ENTRY OF COMMAND MODULE						
		● EARTH LANDING		0.5 hrs.	175.6			
		● RECOVERY OF COMMAND MODULE & CREW						

LUNAR ORBIT RENDEZVOUS MISSION OPERATION PLAN – MISSION CHARACTERISTICS

PREFACE

The Lunar Orbit Rendezvous mode for accomplishing the Apollo manned lunar landing has been studied by the Chance Vought Astronautics Division under contract to Office of Systems, Manned Flight, NASA Headquarters. The objective of this study was to make a systematic and thorough analysis of the Lunar Orbit Rendezvous Mission (LOR) with the end products to be (1) a recommended LOR mission, (2) a recommended vehicle design, and (3) a development plan for accomplishing the overall mission. The study was performed under the title, "Apollo Rendezvous Simulator Study, Contract NASw-413" and is classified CONFIDENTIAL.

The study results are presented in two parts:

Part 1 - SUMMARY REPORT - An overall summary of the significant results of the study.

Part 2 - A complete TECHNICAL DATA REPORT in eight volumes.

Volume I	MISSION SUMMARY AND TRAJECTORY ANALYSIS
Volume II	VEHICLE ANALYSIS - DESIGN
Volume III	VEHICLE ANALYSIS - PROPULSION
Volume IV	VEHICLE ANALYSIS - CONTROLS AND ELECTRONICS
Volume V	VEHICLE ANALYSIS - CREW INTEGRATION AND SAFETY
Volume VI	VEHICLE ANALYSIS - ENVIRONMENTAL CONTROL SYSTEM
Volume VII	VEHICLE ANALYSIS - WEIGHT ANALYSIS
Volume VIII	DEVELOPMENT PROGRAM

The study was conducted within the overall program philosophy and constraints included in the NASA contract statement of work for this study. The principal constraints established in this statement of work are as follows:

- No changes in the Apollo spacecraft design are expected from the result of this study.

- No changes in the Saturn C-5 launch vehicle configuration are expected from the result of this study.

In addition to the contract statement of work, NASA Headquarters defined a series of guidelines for the conduct of the study which were summarized in "Minutes of Lunar Orbit Rendezvous Meeting, April 2 - 3, 1962". The principal guidelines established by this document are as follows:

- The Lunar Excursion Module (LEM) will have a point landing (±1/2 mile) capability.
- The LEM will have redundant guidance and control for each phase of the lunar maneuvers.
- Both automatic and manual guidance and control systems are to be considered in this redundant capability.
- Radio aids, including use of a beacon and/or transmitter on the lunar surface to provide a completely automatic landing are to be studied.
- The suggested hover capability for the LEM is one minute at 100 ft. altitude plus 45 seconds of translation time over the lunar surface. This requirement will be studied further.
- The LEM shall include two crew members.
- The LEM should have a pressurized cabin which has a capability for a one week operation.
- Access to the LEM from Apollo during the earth-moon phase shall be possible.
- The possibility of keeping the LEM attached to the spacecraft on the return moon-earth phase shall be considered.

In general, the philosophy and guidelines established for the study required an examination of all of the important possibilities and techniques for accomplishing the mission. The resulting recommended mission and vehicle design are therefore more comprehensive than a minimum mission and vehicle. In addition to this recommended vehicle, data are presented showing the effect on vehicle weight of mission and design parameters such as lunar stay time, number of crew members, etc.

This volume presents the results of the main propulsion and reaction control system parametric studies. A description of the selected configuration and a recommended development plan for each system are included.

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6.0 MAIN PROPULSION SYSTEM

6.1 INTRODUCTION

The objective of this study was to define the best possible propulsion system for the Lunar Excursion Module assuming an operational time period of 1965 to 1966.

Upon initiation of the study, major emphasis was placed on the collection of propulsion system data. In this survey, the following subjects were carefully covered:

(a) Propellants

(b) Engine performance

(c) Rocket engine components

(d) Propellant storage and pressurization systems

These data were collected from many sources, however, the largest percentage was provided by the various rocket engine manufacturers. Significant contributions were made by Aerojet-General, Bell Aerosystems, Pratt and Whitney, Rocketdyne and Thiokol.

All the available data were compared and evaluated. These results permitted the selection of basic information, both specific and parametric, for use in the propulsion system studies. With these data, systems using both earth storable and cryogenic propellants were synthesized and evaluated.

The first section of this report describes the propulsion system parametric studies performed and presents the results obtained. Here, consideration was given to propellant combinations, vehicle staging, number of engines, thrust to weight ratio, etc., and their effect on total LEM weight.

In the following section, the recommended propulsion system configuration is selected utilizing data from the previous section. In the subsequent section, a detailed definition of the selected system, along with a description of its operation, is provided.

The last section provides a propulsion development plan believed to be realistic and consistent with the LEM time schedule.

To avoid undue complications in discussing the basic studies and the system selection, certain specific and parametric data, along with sample calculations, have been included in appendix form. These are entitled:

(a) Propellant Selections

(b) Parametric Engine Systems Data

(c) Propellant Storage and Pressurization Systems

(d) Rocket Engine Components

(e) Sample Calculations for Propulsion System Evaluation

6.2 SUMMARY

Parametric studies conducted during this study program showed that the propulsion system required to accomplish the Lunar Orbit Rendezvous Mission can be developed in sufficient time for a 1965-1966 mission. In addition, the study showed that the Apollo total weight with either an earth storable or a cryogenic LEM propulsion system is compatible with the Saturn C-5 payload capability.

In selecting the recommended LEM propulsion system,the following criteria were considered: (a) mission success, (b) crew safety, (c) development time, and (d) general compatibility of the vehicle with the entire Apollo system. The propulsion system selected is summarized below:

Propellants	- Earth storables Nitrogen tetroxide and Aerozine-50
Propellant Feed System	- Pressure feed system using cold helium gas, stored at 3000 psi
Vehicle Staging	- 1-1/2 - stage (empty tanks and pressurant system staged)
Engine	- Three engines, ablative thrust chambers, fixed area injectors, gimbaled engines

Earth storable propellants were selected because this selection offered the minimum possible system development time. The helium cold gas pressurization system was selected on the basis of its simplicity and reliability. A weight advantage as well as a more desirable vehicle arrangement resulted from the choice of the 1-1/2 stage concept over the conventional 2-stage tandem concept.

The selected engine system utilizes gimbaling with three engines to achieve both throttle control (with a fixed injector) and engine redundancy. The following interesting observations were made with respect to the engine system: A gimbaling system provides the most practical method for achieving engine redundancy. Once the gimbaling system is introduced, it affords a technique which can be used to achieve throttle control in multiple engine installations with no basic increase in weight or complexity. Using a three engine arrangement, effective redundancy and throttling can be achieved with the relatively simple fixed area injector engines (with upstream throttling). Whether engine redundancy is required may be open to question. However, considering the development status of ablative chambers along with the susceptibility of regeneratively cooled chambers to damage, the safest approach appears to be redundancy.

Ablative chambers were selected for the recommended system because of their inherent ruggedness, and because the development time involved is less than for regeneratively cooled chambers. In addition, the cooling capacity available with storable propellants limits the throttle ratio to approximately 3 to 1. It is possible that regenerative cooling could be used with a 3-engine storable propellant system, but it is not possible to use regenerative cooling with a single engine system. In either case, a clear preference is indicated for the ablative system.

The program and development time data obtained from the engine manufacturers indicate that a Pre-flight Rating Test engine can be available in 17 months from contract go-ahead and that a fully man-rated engine can be achieved in 33 months from go-ahead. Delivery of operational engines could begin as early as 20 months from program go-ahead.

6.3 PROPULSION SYSTEM PARAMETRIC STUDIES

The parametric studies conducted to define the most practical propulsion system for the LEM were divided into two phases: (a) Preliminary Parametric Evaluation and (b) Final Parametric Evaluation. Phase (a) was conducted early in the study program and prior to the time when specific vehicle data became available. This phase provided a first approximation to the type of propulsion system required. Phase (b) was conducted after specific vehicle data were available and consisted of a thorough analysis with the objective of defining the most practical propulsion system for the LEM. These phases are discussed separately in the following paragraphs.

6.3.1 Preliminary Parametric Evaluation

The purpose of this portion of the study was to obtain a first approximation of the type of propulsion system required for the LEM mission. This approximation was required to provide guidance for more detailed system investigations by all study areas.

In this evaluation various propulsion systems were synthesized using state of the art data consistent with the predicted LEM operational time period. The data utilized consisted of: (a) propellant characteristics, (b) parametric engine data, (c) preliminary propellant storage and pressurization data, and (d) rocket engine component characteristics. The final form of these data, along with appropriate discussions, is presented in Appendices 6A through 6D.

Based on the information provided in the Appendices mentioned above, several systems were selected for preliminary evaluation. Tables 6-I and 6-II show the reasoning employed in selecting the systems to be evaluated. In these tables, the shaded items are considered to represent areas of minimum technical risk for the operational time period of the LEM.

From these tables, study emphasis was indicated for systems utilizing the following:

(1) Storable propellants

- Ablative thrust chambers
- Pressure feed system
- Cylindrical propellant tanks

(2) Cryogenic propellants

- Ablative thrust chambers
- Pressure feed system
- Cylindrical propellant tanks

(3) Cryogenic propellants

- Regenerative thrust chamber
- Pump feed system
- Cylindrical propellant tanks

TABLE 6–I PRELIMINARY PROPULSION PARAMETRIC STUDY CONSIDERATIONS (STORABLE PROPELLANTS)

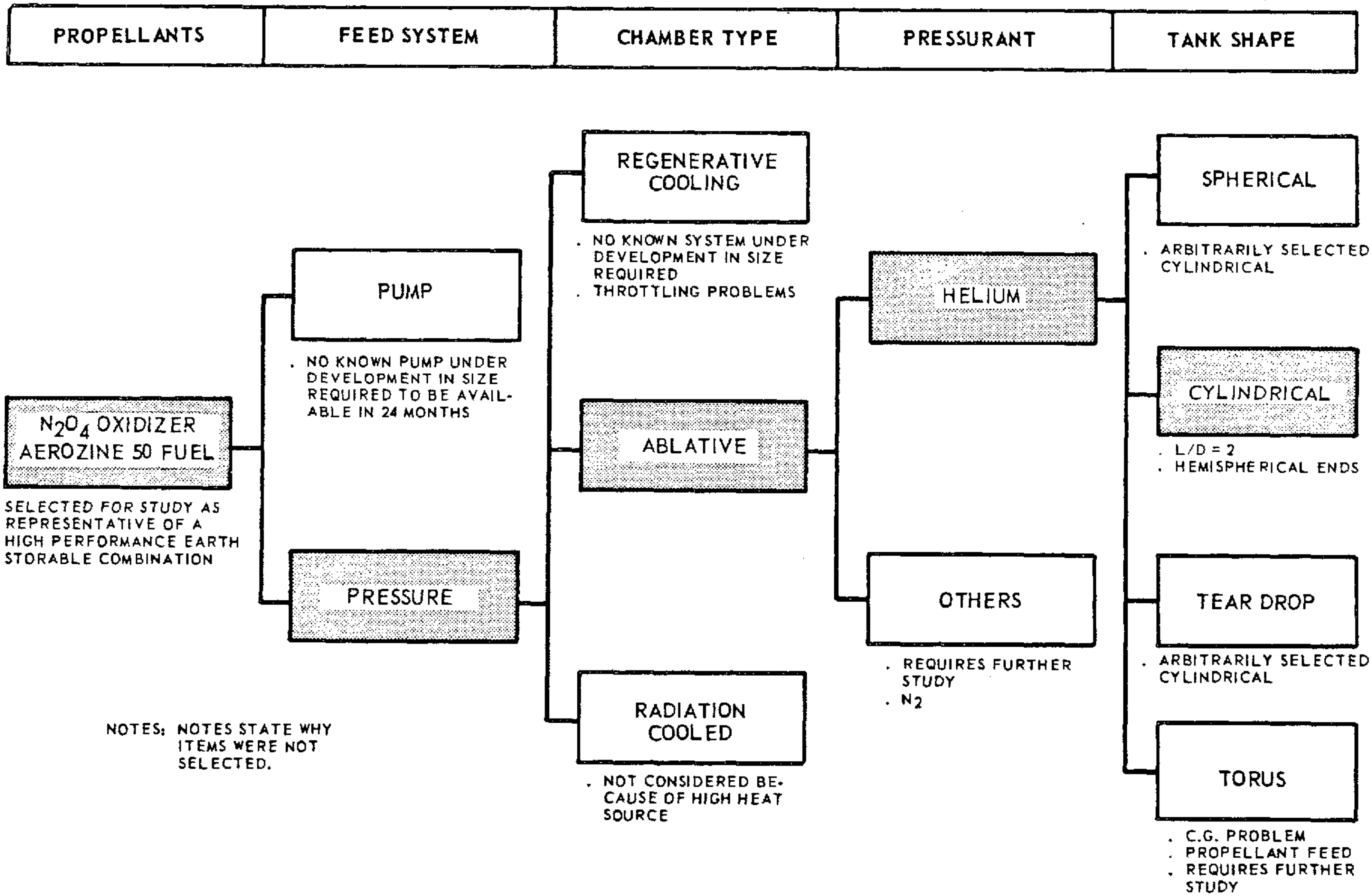
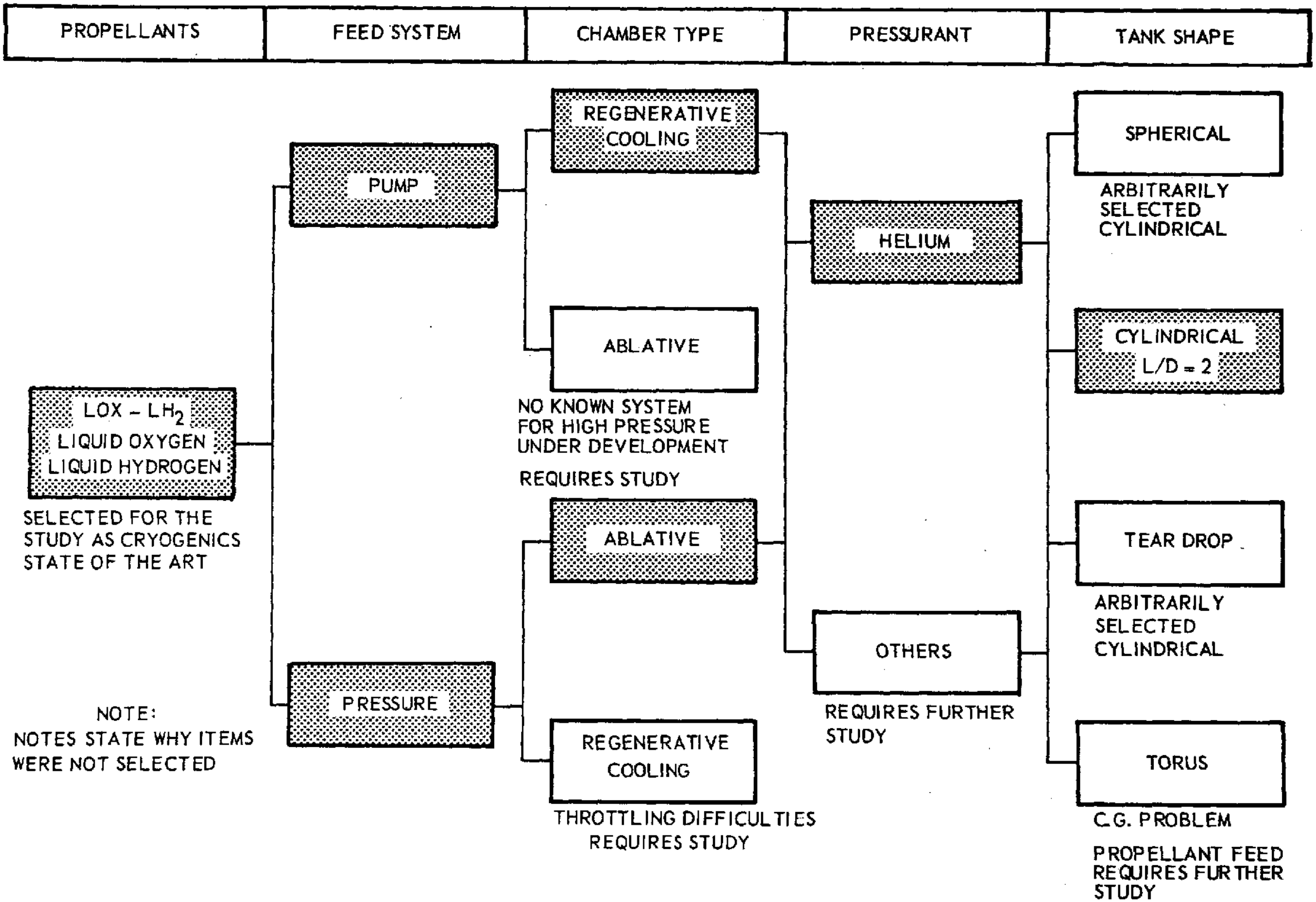


TABLE 6–II PRELIMINARY PROPULSION PARAMETRIC STUDY CONSIDERATIONS (CRYOGENIC PROPELLANTS)



Using the propulsion systems defined by these combinations, several vehicle arrangements were evolved for evaluation. These arrangements were selected to provide information on the effect of: (a) propellants, (b) staging, and (c) multiple thrust chambers. The storage propellants selected for evaluation were nitrogen tetroxide and Aerozine-50. Liquid oxygen and liquid hydrogen were selected as the cryogenic propellant for the evaluation. The staging concepts considered involved that of staging empty tanks in one case and that of staging both engines and tanks in the other. In the latter case, the conventional stage arrangement was considered, i.e., the ascent stage was independent of the descent stage and physically located above the descent stage. Some definite advantages accrue from staging the vehicle before lunar landing, however, because of the safety problems foreseen in following this procedure, only staging on the lunar surface was considered in this study. Multiple chamber configurations were considered in this evaluation to indicate the penalty for redundant engines. A four thrust chamber design was selected because it easily adapts itself to symmetrical thrust control in case of an engine failure.

A definition of each of the systems investigated is provided in Table 6-III. The method used in evaluating these systems is illustrated by sample calculations in Appendix 6E-1.

The results of the evaluation are presented in Figures 6-1 through 6-4. The most significant fact indicated here is that either storable or cryogenic propellant systems are adequate for use with the LEM when the allowable initial weight (lunar orbit weight) is 30,000 lbs. and the capsule weight does not exceed 6,000 lbs. However, these criteria are met only by the single stage (tank staging only), single chamber arrangement. The single stage, single chamber evaluation results are shown in Figure 6-I. The results also indicate that staging of propulsion systems, if accomplished in the conventional manner, will result in a total system weight penalty. The above results, although of a preliminary nature, were used as guidelines for further study.

6.3.2 Final Parametric Evaluation

The purpose of this phase of the study was to provide a more accurate evaluation of the various types of propulsion systems than was possible in the preliminary studies. This step was possible only after improved data became available from each of the various study areas. These improved data included primarily better weight estimates, additional configuration data and improved estimates of mission energy requirements (ideal velocity change, ΔV). The major weight changes resulted from: (a) establishing the weight of the crew station, and (b) a better definition of the propellant reserve requirements. Perhaps the most significant of the weight changes is that represented by propellant reserve requirements. In preliminary studies this value was established at 2 per cent of the total propellant weight. In this phase of the study, the propellant reserve was established at 10% for descent and 5% for ascent. These increased reserve allowances are explained in Paragraph 17.5.4, Volume II.

The preliminary propulsion system evaluation study indicated a weight advantage for the configuration utilizing lunar staging of tanks and associated components. This conclusion was verified by parallel detail design studies.

TABLE 6-III SUMMARY OF PROPULSION SYSTEMS – PRELIMINARY PARAMETRIC EVALUATION

SYSTEM NUMBER	PROPELLANT TYPE	STAGING ARRANGEMENT	NOMINAL VACUUM SPECIFIC IMPULSE	NUMBER OF THRUST CHAMBERS	PROPELLANT FEED SYSTEM	TYPE OF THRUST CHAMBER	PROPELLANT TANK PRESSURE	THRUST CHAMBER DESIGN PRESSURE	REDUNDANCY
1	N_2O_4– AEROZINE 50	TANKS AND LANDING GEAR STAGED	320	1	PRESSURE FED FROM 6000 PSIA HELIUM SOURCE	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS ETC.
2	N_2O_4– AEROZINE 50	** FIRST STAGE PROPULSION SYSTEM AND LANDING GEAR STAGED	320	2 (1 PER STAGE)	PRESSURE FED FROM 6000 PSIA HELIUM SOURCE	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, ETC.
3	N_2O_4 AEROZINE 50	TANKS AND LANDING GEAR STAGED	320	4	PRESSURE FED FROM 6000 PSIA HELIUM SOURCE	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, PLUS 2 REDUNDANT THRUST CHAMBERS
4	N_2O_4 AEROZINE	** FIRST STAGE PROPULSION AND LANDING GEAR STAGED	320	8 (4 PER STAGE)	PRESSURE FED FROM 6000 PSIA HELIUM SOURCE	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS PLUS 2 REDUNDANT THRUST CHAMBERS PER STAGE
5	*LO_2 – LH_2	TANKS AND LANDING GEAR	420	1	PRESSURE FED FROM 6000 PSIA HELIUM SOURCE	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, ETC.
					PUMP FED	REGENERATIVE	50 PSIA	300 PSIA	ALL DYNAMIC COMPONENTS ETC. EXCEPT PUMP
6	*LO_2 – LH_2	** FIRST STAGE PROPULSION SYSTEM AND LANDING GEAR STAGED	420	2 (1 PER STAGE)	PRESSURE FED	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, ETC.
					PUMP FED	REGENERATIVE	50 PSIA	300 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES REGULATORS, ETC. EXCEPT PUMP
7	*LO_2 – LH_2	TANKS AND LANDING GEAR STAGED	420	4	PRESSURE FED	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, ETC. PLUS 2 REDUNDANT THRUST CHAMBERS
					PUMP FED	REGENERATIVE	50 PSIA	300 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS ETC. PLUS 2 REDUNDANT THRUST CHAMBERS
8	*LO_2 – LH_2	** FIRST STAGE PROPULSION AND LANDING GEAR STAGED	420	8 (4 PER STAGE)	PRESSURE FED	ABLATIVE	200 PSIA	100 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS ETC. PLUS 2 REDUNDANT THRUST CHAMBERS PER STAGE
					PUMP FED	REGENERATIVE	50 PSIA	300 PSIA	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, ETC. PLUS 2 REDUNDANT THRUST CHAMBERS PER STAGE

* SYSTEM EVALUATED WITH BOTH PUMP AND PRESSURE FEED
** LUNAR ASCENT STAGE LOCATED ABOVE DESCENT STAGE

NOTES:
1. STAGING ONLY ON LUNAR SURFACE
2. ΔV DESCENT = 6,500 FPS
   ΔV ASCENT = 6,850 FPS
3. THRUST CHAMBER: EXPANSION RATIO 40 TO 1
   MINIMUM BURN TIME 10 MIN. THRUST RANGE 5000 TO 15000 LB.
4. WEIGHT BASED ON 2% UNAVAILABLE PROPELLANT
5. STORABLE STORAGE TEMPERATURE 40 TO 100° F
6. NON VENTED CRYOGENICS SYSTEMS
7. HELIUM STORAGE TEMPERATURE
   STORABLES – 100° F
   CRYOGENICS – CRYOGENIC H_2 TEMPERATURE
8. STRUCTURE = 5% LOADED WEIGHT
9. LANDING GEAR = 5% LUNAR LANDED WEIGHT
10. CRYOGENIC TANKAGE INSULATION:
    MULTILAYER INSULATION ($K = 2 \times 10^{-5}$ BTU –FT/HR–FT^2°R
    OUTER SURFACE TEMPERATURE = 400°R FOR 100 HRS.
    = 200° FOR NEXT 200 HRS.
11. CYLINDRICAL TANKS L/D = 2 HEMISPHERICAL ENDS
    STRENGTH TO DENSITY RATIO (σ/ρ = 350,000 INS) 1.5 OFF YIELD, 1.995 OFF ULTIMATE @ 100° F
    20% CONTINGENCY FOR FITTINGS, MFG. TOLERANCE, ETC.

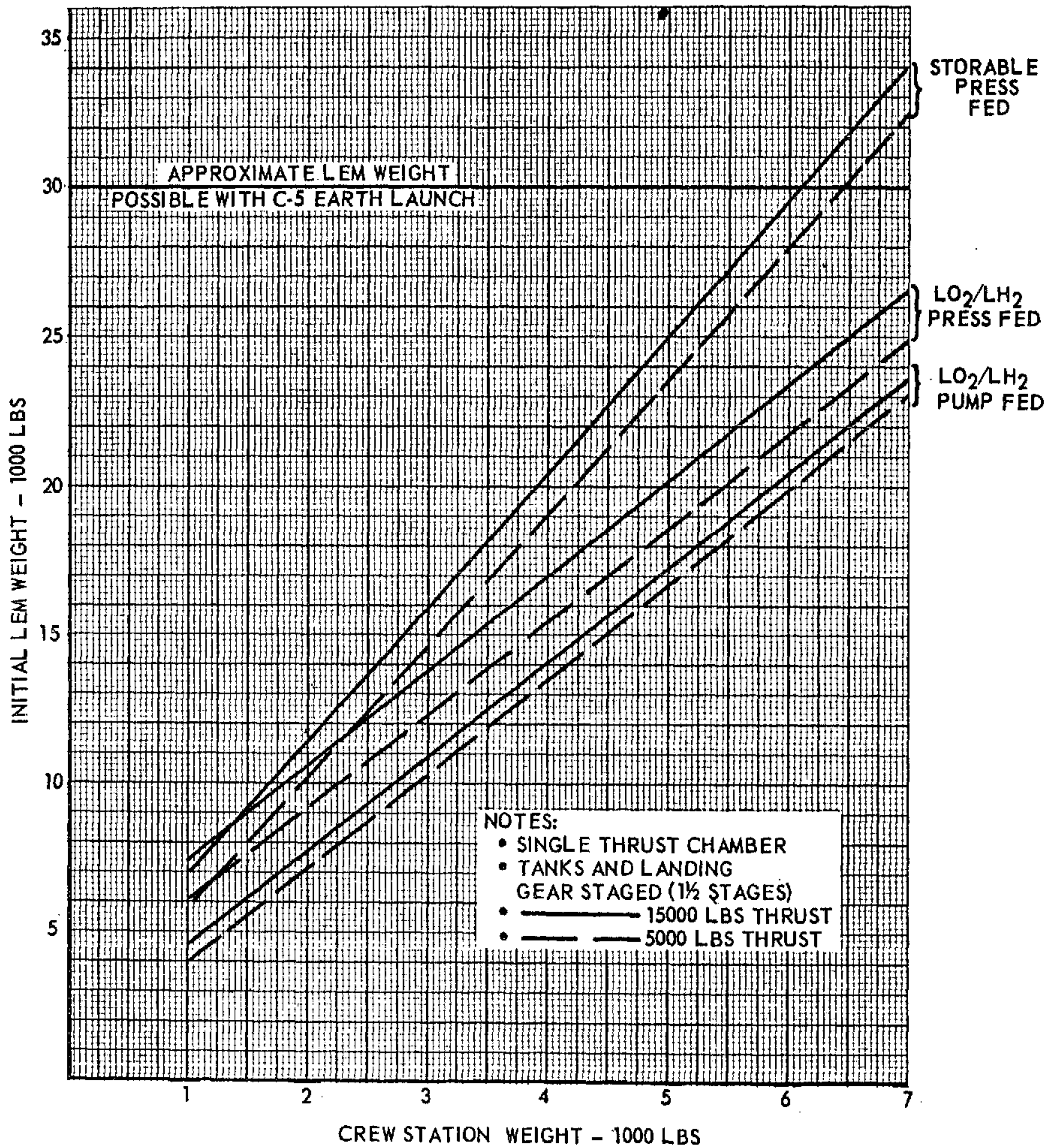


Figure 6-1 VARIATION IN INITIAL LEM WEIGHT WITH CREW STATION WEIGHT (SINGLE ENGINE - 1½ STAGES)

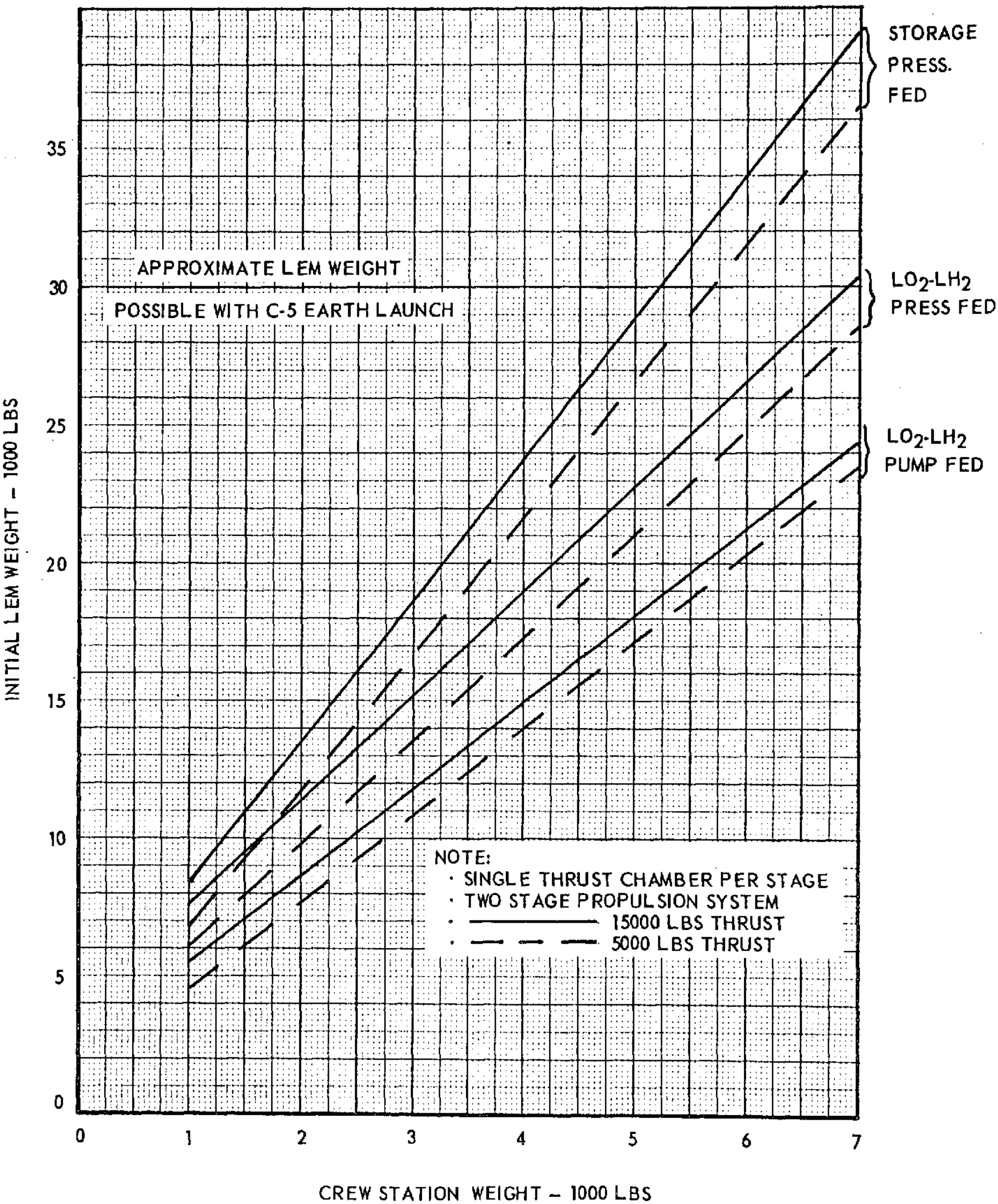


Figure 6–2 VARIATION IN INITIAL LEM WEIGHT WITH CREW STATION WEIGHT (SINGLE ENGINE – 2 STAGES)

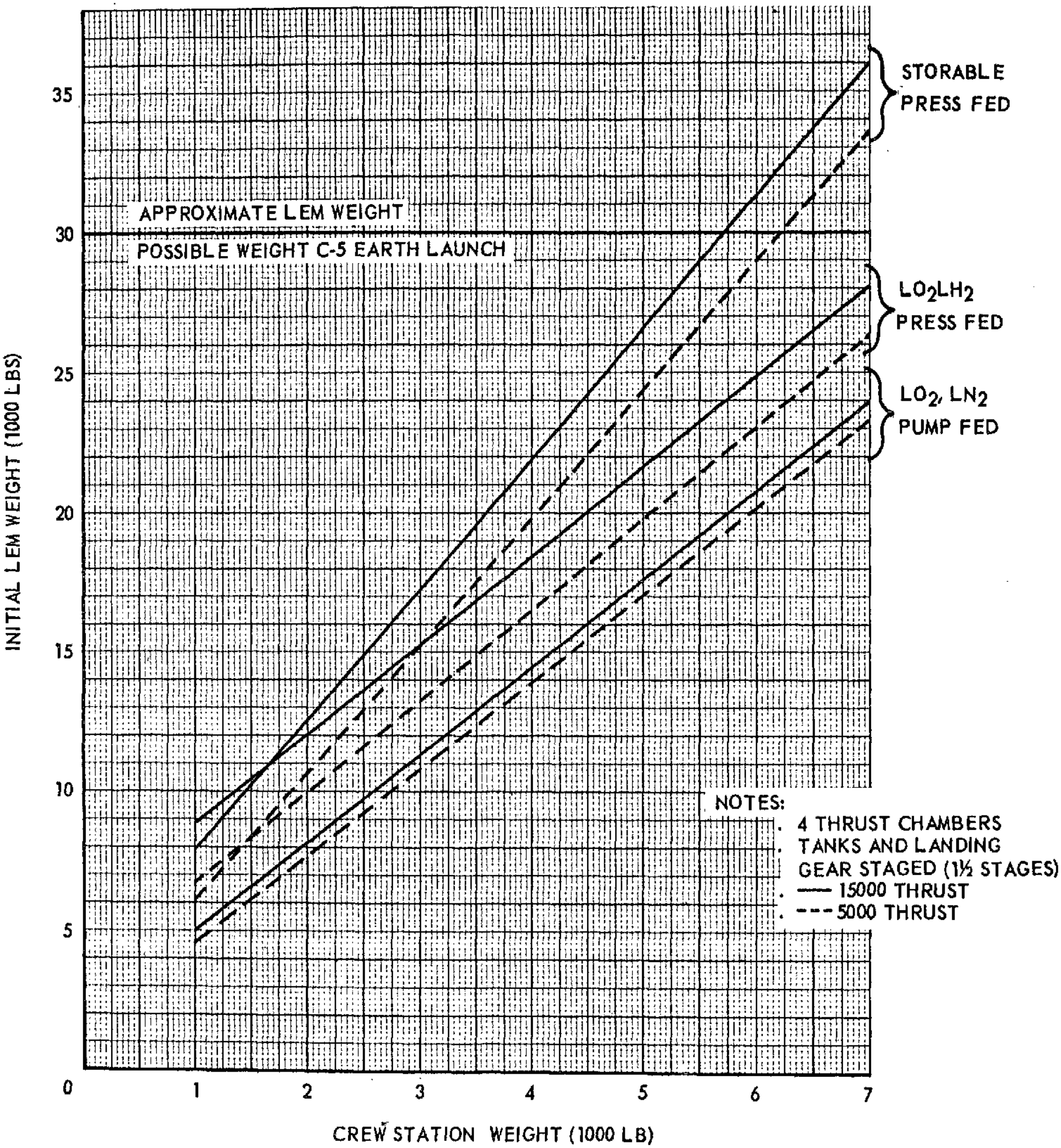


Figure 6-3 VARIATION IN INITIAL LEM WEIGHT WITH CREW STATION WEIGHT (FOUR ENGINES – 1½ STAGES)

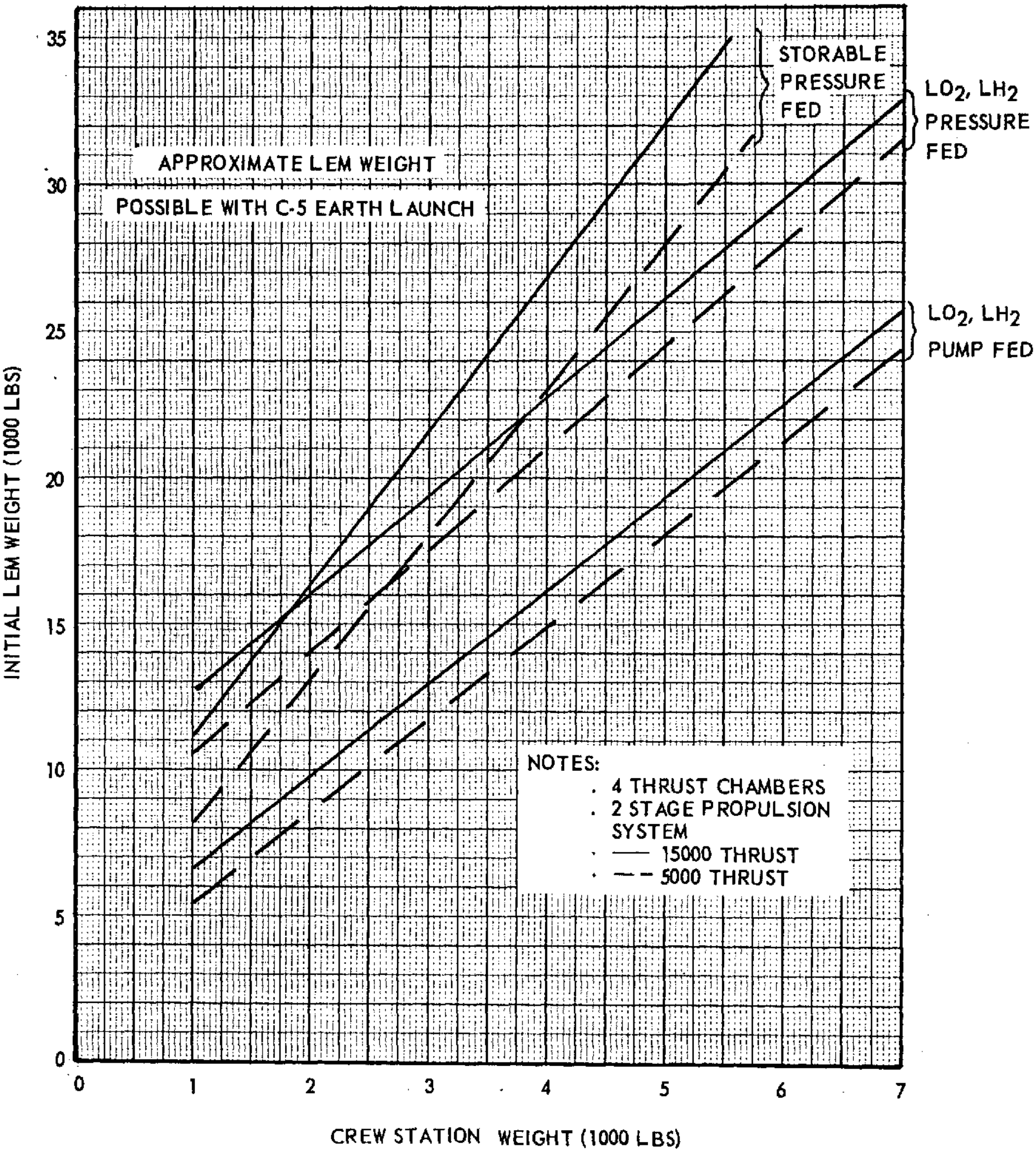


Figure 6–4 VARIATION IN INITIAL LEM WEIGHT WITH CREW STATION WEIGHT (FOUR ENGINES – 2 STAGES)

The most important new input provided to the study at this point was parametric data on mission energy requirements. These requirements are established by the descent and ascent trajectories followed plus the energy required for the following maneuvers:

- Retro for lunar descent
- Hover
- Translation and letdown
- Plane change for rendezvous with the command module

Assuming a given guidance system, the energy requirements become a function of lunar orbit altitude, vehicle thrust to weight ratio and the maneuvers performed. The inter-relationship of the orbit altitude and thrust to weight ratio is shown in Figures 6-5 and 6-6 for the descent and ascent trajectories, respectively.

The maneuver energy requirements established are tabulated below.

Maneuver	Ideal Velocity Change, ΔV
Retro for lunar descent	255 ft/sec.
Hover	320
Translation and letdown	270
Plane change	25

The data presented in Figures 6-5 and 6-6 indicate that for each lunar orbit altitude of operation, the minimum energy required is associated with a unique thrust to weight ratio. In the case of descent, this value varies from 0.8 for an orbit of 100,000 ft. to a value of 1.0 for a 50,000 ft. orbit. For the ascent phase of the mission, the optimum thrust to weight ratio, based on minimum energy requirements, is almost independent of altitude and has a value of approximately 0.7. If energy requirements were the only consideration, the thrust to weight ratio for optimum vehicle weight could be established from these data. This results from the fact that the energy requirement (ideal velocity change, ΔV) is almost a direct function of propellant weight, which represents a large percentage of the vehicle weight. However, the data presented in Figures 6-5 and 6-6 are based entirely on trajectory considerations and do not include the effect of propulsion system weight. Therefore, before actual optimization can be achieved, these weights must be introduced.

Items which make up the propulsion system weight may be divided into two categories: (a) weight which is dependent upon propellant weight, and (b) weight which is dependent upon engine thrust level. The propellant dependent weight items consist of: propellant tanks, tank insulation, pressurant, pressurant system, etc. For the purpose of this study, these component weights have been defined and are presented in Appendix 6C. Thrust dependent weight items consist of: thrust chambers, control valves, regulators, gimbal systems, etc., and are defined and presented in Appendix 6B.

By using these data in conjunction with the energy requirement (ideal velocity change, ΔV) data, it is possible to establish the optimum

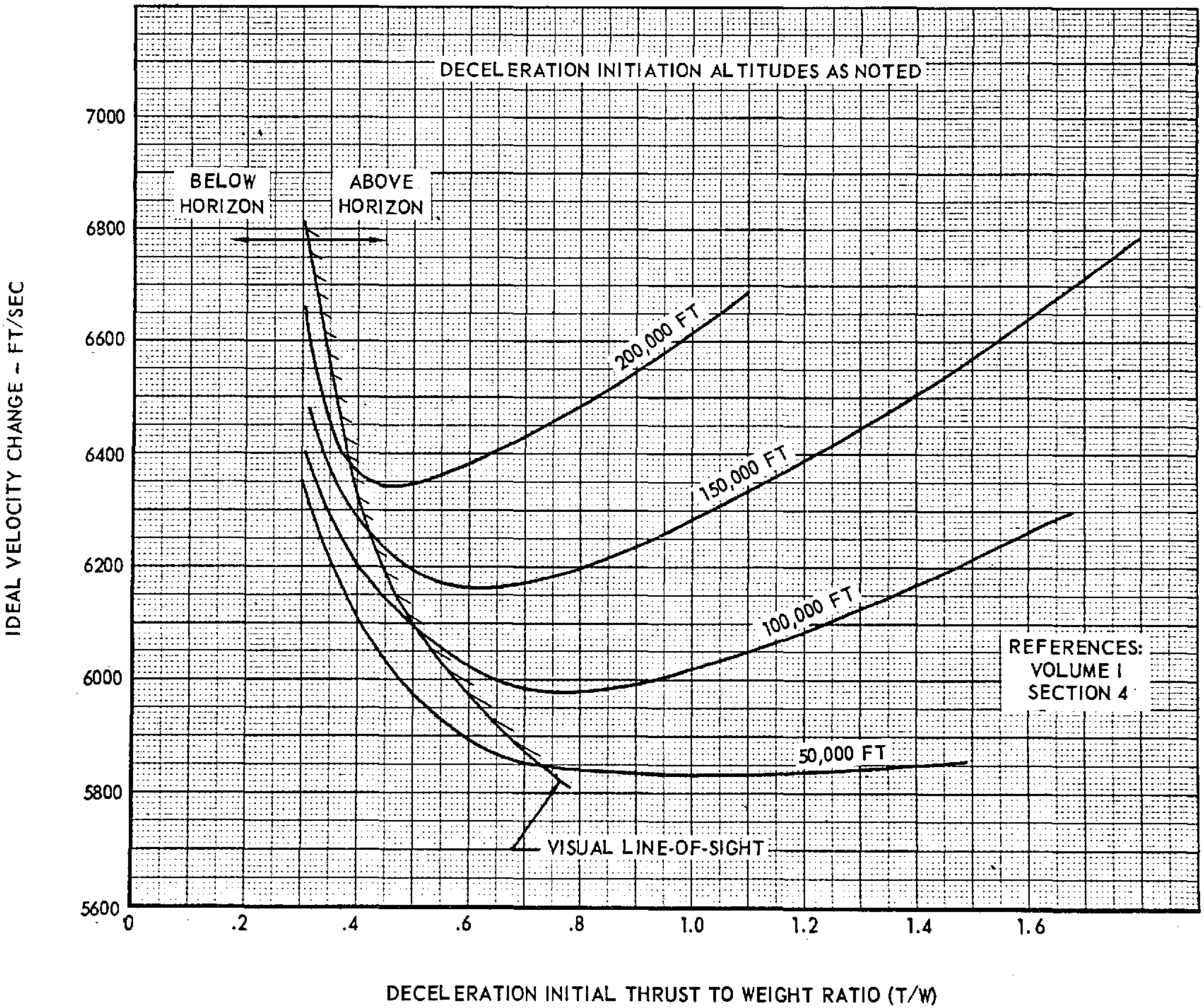


Figure 6-5 IDEAL VELOCITY CHANGE TO DECELERATE

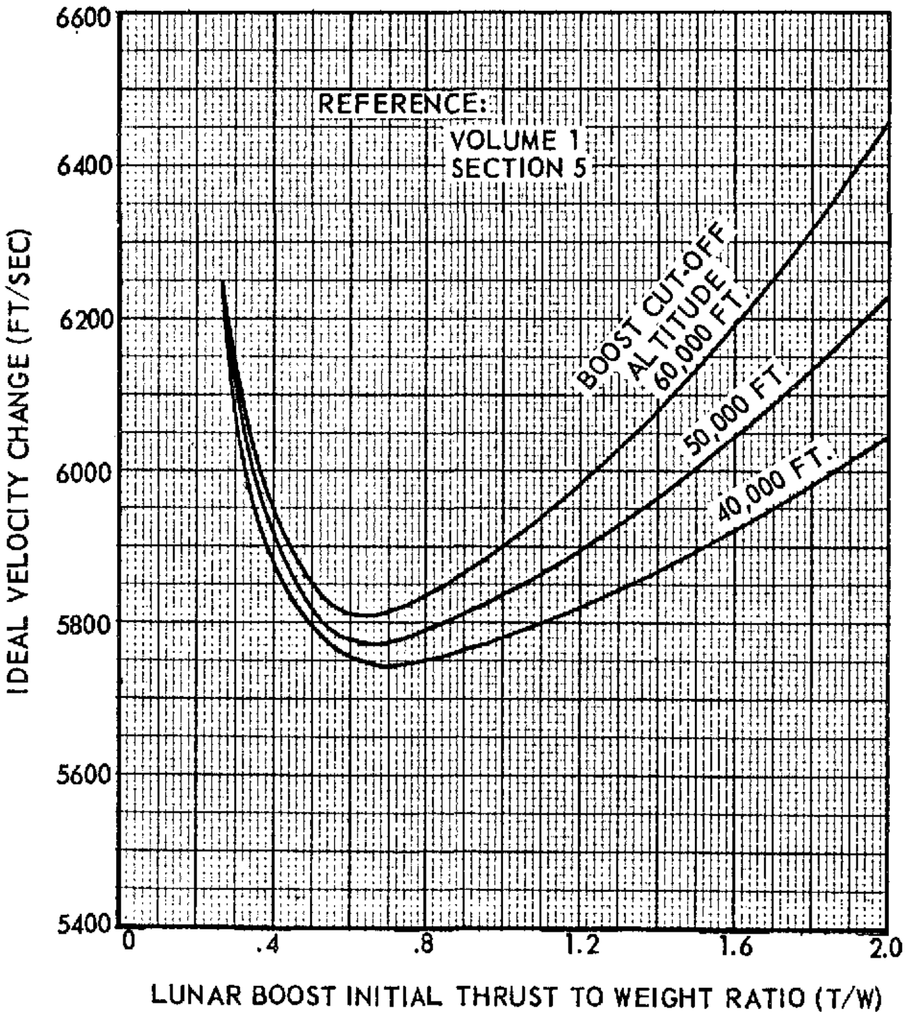


Figure 6-6 LUNAR BOOST – IDEAL VELOCITY CHANGE

thrust to weight ratio along with the minimum weight of any given vehicle configuration. Five vehicle configurations were selected for evaluation in this phase of the study. Perhaps, the most important factor considered in making these configuration selections is the requirement for engine throttling. This requirement becomes evident upon examination of the energy requirements (ideal velocity change, ΔV) data for descent. Figure 6-6 indicates that a thrust to weight ratio, at initiation of descent, in the order of 0.7 or more may be required for optimum design. Figure 6-7, which is based on a typical descent mission, shows that for an initial thrust to weight ratio of 0.7, a throttle ratio of the order of 10 to 1 is required. Therefore, each of the systems selected for this evaluation, except one, have a throttling capability of 10:1 or greater.

Engine throttling may be accomplished in one of three ways: (a) flow control of propellants with an upstream control value, (b) flow control of propellant by varying the injector area at the thrust chamber face, or (c) control of propellant feed pump speed.

The throttle ratio possible using upstream flow control is very limited. Analyses show that, because of high pressure losses in the fixed area thrust chamber injector, the maximum practical throttle ratio is approximately 3.5 to 1. For ratios above this value, the propellant tank pressures required cause prohibitive increases in system weight. However, an effective throttle ratio of 10 to 1 can be realized by utilizing 3 thrust chambers, with provision for operating any combination. Upstream throttling is within the current state of the art. Injector characteristics and performance data for upstream throttling are reported in Appendix 6B.

By use of flow control at the thrust chamber face (variable area injector), a throttle ratio in excess of 10 to 1 can be easily achieved without a penalty in tank weight. Unfortunately, variable area injectors are not considered to be within the current state of the art. A discussion of this subject is provided in Appendix 6B.

Pump fed, regeneratively cooled thrust chambers can be operated over a throttle ratio of 10 to 1. However, because of the limited cooling capacity available with storable propellants, throttle ratios in excess of 3 or 4 are not practical because of nozzle overheating. These characteristics of pump fed regenerative systems are discussed in Appendix 6B.

The basic characteristics of the propulsion systems selected for evaluation are as follows:

1. Storable Propellants

    - Single engine
    - Fixed injector
    - Pressure feed system
    - Ablative thrust chamber

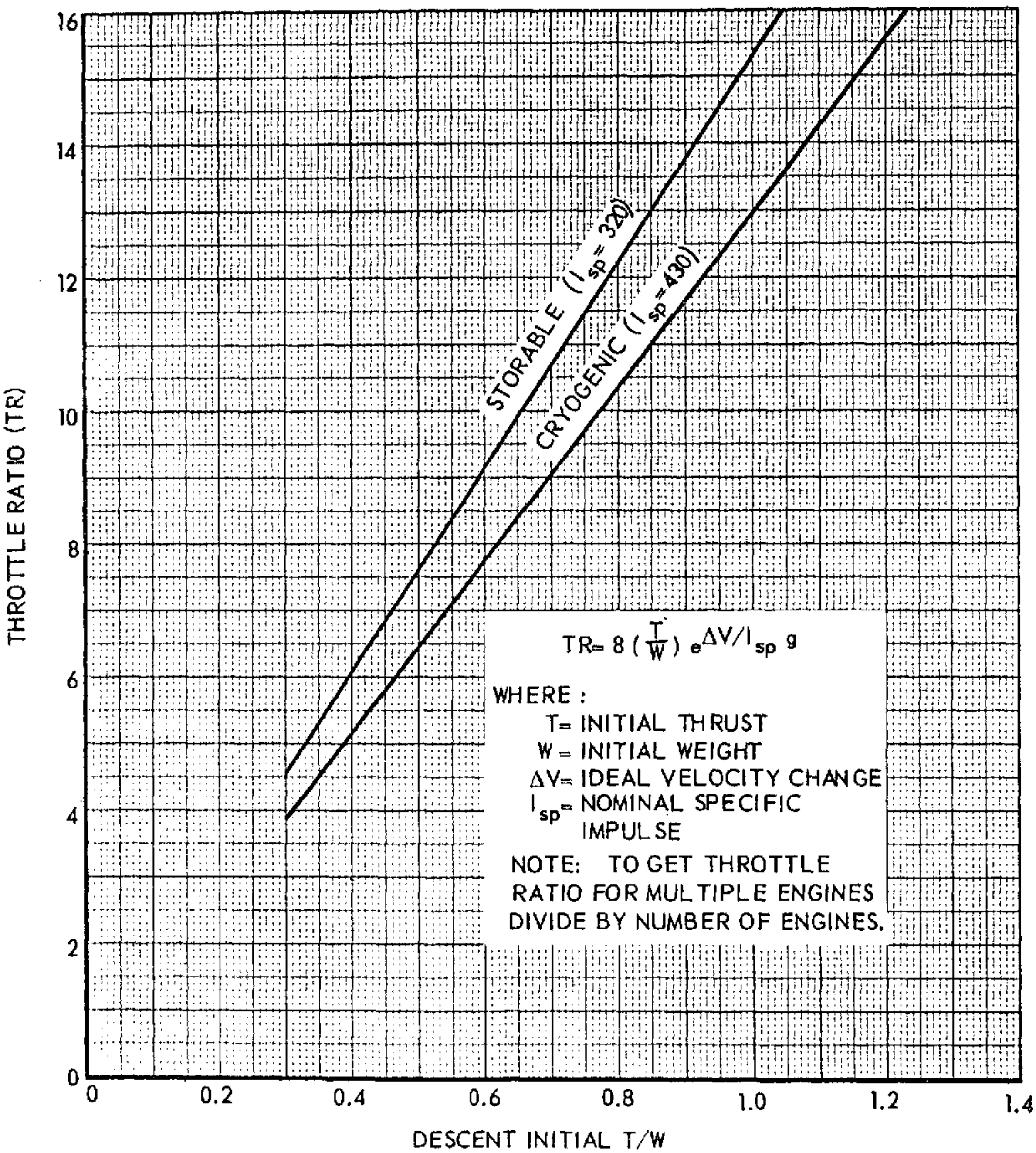


Figure 6-7 REQUIRED THROTTLE RATIOS VS. DESCENT T/W (TYPICAL)

2. Storable Propellants

- Three engines
- Fixed injectors
- Pressure feed systems
- Ablative thrust chambers

3. Storable Propellants

- Single engine
- Variable injector
- Pressure feed system
- Ablative thrust chambers

4. Cryogenic Propellants

- Three engines
- Fixed injectors
- Pressure feed system
- Regeneratively cooled thrust chambers

5. Cryogenic Propellants

- Single engine
- Fixed injector
- Pump feed system
- Regeneratively cooled thrust chamber

Table 6-IV defines the above propulsion systems in greater detail, and also provides the vehicle data used in performing the vehicle evaluation. The method used in conducting these evaluations is illustrated by a sample calculation in Appendix 6E, Paragraph 2.0.

The results of these evaluations are well summarized in Figure 6-8. The variation of LEM initial weights for each configuration is provided for a wide range of initial thrust to weight ratios. Also, for three of the configurations, the effect of orbit altitude on vehicle weight is included. Altitude is important in selecting the vehicle thrust to weight ratio, since line-of-sight between the LEM and the selected lunar landing site is assumed to be required. To show the relationship between thrust to weight ratio and altitude, a horizon line is superimposed on the parametric altitude curves. In these plots, points above the horizon line allow line-of-sight operation while points below do not. The relative flatness of the line-of-sight curve indicates that the LEM weight is almost independent of thrust to weight ratio over a wide range. Therefore, if line-of-sight operation is required, it appears that the thrust to weight ratio should be chosen as high as possible to permit system growth.

Perhaps the most significant conclusion which can be drawn from this evaluation is that the gross weight of each of the systems analyzed, except one, is within the payload capability (LEM - 30,000 lbs.) of the Saturn C-5 launch vehicle. The excessive weight of the unacceptable system, designated Configuration (1) in Figure 6-8, results primarily from the lack of

TABLE 6-IV

SUMMARY OF PROPULSION SYSTEMS – FINAL PARAMETRIC STUDIES

SYSTEM NO.	PROPELLANT TYPE	NOMINAL Isp	NO. OF ENGINES	PROPELLANT FEED SYSTEMS	TYPE OF THRUST CHAMBER	PROPELLANT TANK PRESSURE	THRUST MAX. CHAMBER PRESSURE	TYPE INJECTOR	REDUNDANCY
1.	N_2O_4 AND AEROZINE 50	320	1	PRESSURE FEED FROM 3000 PSI HELIUM SOURCE	ABLATIVE	VARIES WITH THROTTLE RATIO	100 PSIA	FIXED	ALL DYNAMIC COMPONENTS INCLUDING: VALVES, REGULATORS, ETC.
2.	N_2O_4 AND AEROZINE 50	320	3	PRESSURE FEED FROM 3000 PSI HELIUM SOURCE	ABLATIVE	VARIES WITH THROTTLE RATIO	100 PSIA	FIXED	ALL DYNAMIC COMPONENTS INCLUDING: VALVES, REGULATORS, ETC.
3.	N_2O_4 AND AEROZINE 50	320	1	PRESSURE FEED FROM 3000 PSI HELIUM SOURCE	ABLATIVE	130 PSIA	100 PSIA	VARIABLE	ALL DYNAMIC COMPONENTS INCLUDING VALVES, REGULATORS, ETC.
4.	LO_2 LH_2	430	3	PRESSURE FEED FROM 3000 PSI HELIUM SOURCE	REGENER-ATIVE	200 PSIA	100 PSIA	FIXED	ALL DYNAMIC COMPONENTS INCLUDING: VALVES, REGULATORS, ETC.
5.	LO_2 LH_2	430	1	PUMP	REGENER-ATIVE	50 PSIA	300 PSIA	FIXED	1 ENGINE PLUS VALVES, REGULATORS, ETC.

NOTES:
1. ALL VEHICLE STAGE LANDING GEAR, EMPTY PROPELLANT TANKAGE AND PRESSURIZATION SYSTEM ON LUNAR SURFACE.
2. ALL TANKS ARE CYLINDERICAL.
3. NOZZLE EXPANSION RATIO IS 40.
4. CRYOGENIC SYSTEMS ARE UNVENTED.
5. LANDING GEAR IS 5% TOTAL VEHICLE WEIGHT.
6. CREW STATION WEIGHT – 5187

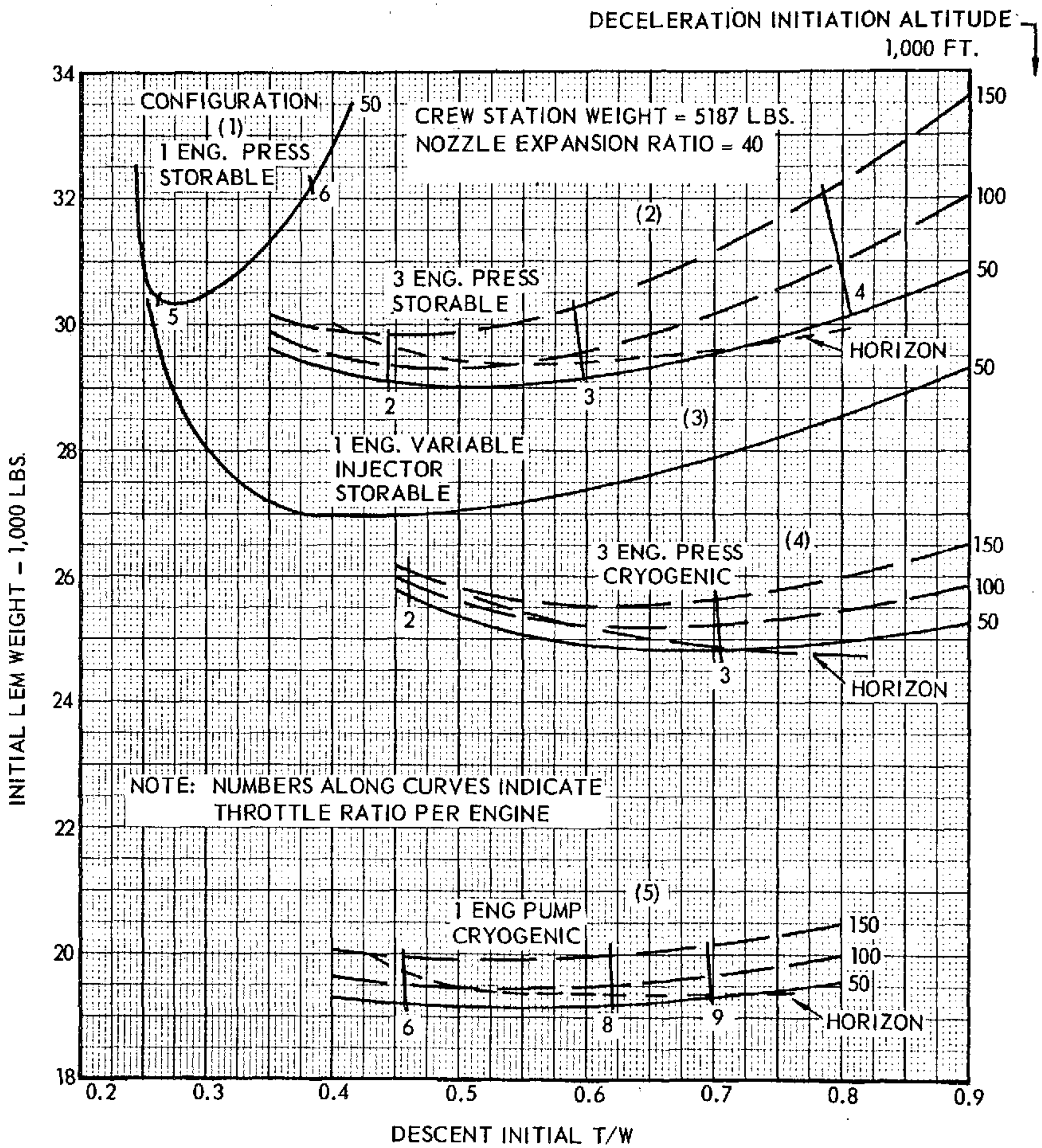


Figure 6-8 PARAMETRIC PROPULSION SYSTEM EVALUATION

throttling capability provided by the engine. Relative to the acceptable configurations, it is significant to note that the development time associated with each vehicle is proportional to the weight levels indicated in Figure 6-8. A minimum of development time will be required for the three engines, pressure fed, storable propellant system, while a maximum of development time is associated with the pump fed, regeneratively cooled cryogenics system.

6.4 FINAL SYSTEM SELECTION

This section presents the method utilized in selecting the recommended LEM propulsion system. It also provides the final system performance and a summary justification for each of the significant systems and components selected.

6.4.1 Engine System Selection

In making this selection, major emphasis was placed on the necessity of achieving: mission success, crew safety, minimum development time, and compatibility of the configuration with the Apollo system. To accomplish these objectives, wherever possible, only components and subsystems with extensive state of the art background were considered. Also, a redundancy philosophy was established which provides for one dynamic part failure without affecting mission success and two such failures without affecting crew safety.

In the preceding parametric studies it was determined that the LOR mission could be accomplished using either storable or cryogenic propellants. However, because minimum development time is associated with storable systems, storable propellants were emphasized for the LEM vehicle.

To aid in selecting the propulsion system with the most desirable characteristics, Table 6-V was prepared. This table presents a number of systems covering a wide range of possible arrangements. Many of these systems do not meet all the requirements established above, but are included in order to provide as complete a coverage as possible. In fact, only one of the systems, designated System No. 2, meets both the redundancy and throttle requirement (indicated in the table). This system, however, was considered unacceptable because it employs a variable area injector, the development of which is incompatible with the LEM program. Also, the vehicle design arrangement required to accommodate the two large engines of System No. 2 is considered undesirable. In fact, detail design studies revealed that, because of landing gear problems and general vehicle compactness, the most attractive vehicle arrangement could be achieved with a 3-engine arrangement. Fortunately, this type arrangement also offers an attractive method of achieving engine redundancy as well as throttle control without major penalties. Four such systems were considered and are included in Table 6-V.

An examination of two arrangements, which consist of three fixed engines and variable injectors (System Nos. 5 and 7), reveals that in the event of engine failure either one can execute an abort in spite of the asymmetrical thrust provided. However, a safe lunar landing for the engine out condition cannot be effected with either of these systems.

The two systems which utilize gimbaling and fixed area injectors (System Nos. 4 and 6) present different problems. Since thrust alignment can be easily accomplished with the gimbal system, this presents no problem. However, the system which has one completely redundant engine (System No. 4) cannot complete a lunar landing because of insufficient throttle ratio (5:1 per engine required). A lunar landing can be successfully effected with

TABLE 6-V ENGINE SYSTEM CONSIDERATIONS

SYSTEM NO.	NO. OF ENGINE	REDUNDANT ENGINE	INJECTOR TYPE	GIMBALED	THROTTLE RATIO REQUIRED/ENGINE
1	1	NO	VARIABLE	NO	10
2	2	YES	VARIABLE	½*	10
3	2	NO	VARIABLE	NO	10
4	3	½*	FIXED	YES	5
5	3	½*	VARIABLE	NO	15
6	3	NO	FIXED	YES	3.3
7	3	NO	VARIABLE	NO	10
8	2-RL-10*	YES	FIXED	½*	7

*NOTES:
1. THE DESIGNATION ½ UNDER THE GIMBALED COLUMN INDICATES THAT THE GIMBAL HAS ONLY ONE DEGREE OF FREEDOM, TO PROVIDE ENGINE REDUNDANCY.
2. THE DESIGNATION ½ UNDER THE REDUNDANT ENGINE COLUMN INDICATES THAT ONLY ONE SPARE ENGINE IS AVAILABLE (TWO ARE OPERATING).
3. ALL ENGINES CONSIDERED USE STORABLES PROPELLANTS EXCEPT THE RL-10.
4. ALL SYSTEMS ARE 1½-STAGES (TANKS AND PRESSURANT SYSTEMS STAGED ON LUNAR SURFACE.

System No. 6 with one engine out, but at a point down range from the planned site. Calculation shows that if engine failure occurs at the beginning of descent, the overshoot distance is approximately 70 miles. This is a very attractive feature which provides a form of redundancy that can be achieved simply by planning an alternate but equally acceptable landing site in case of an engine failure during descent.

It is interesting to note that the gimbaling nozzle is the key to the attractiveness of the 3-engine concept described above. Since with this concept an adequate throttling ratio can be achieved with either variable area injector or fixed area injector engines, the obvious choice is the simpler fixed area injector engine.

Another interesting observation which can be made from the above considerations is that, in order to achieve engine redundancy, a gimbal system must be used. Once this is accepted it appears logical that the gimbal could also be used to achieve throttle control if this simplifies the engine problem. For the LEM application this is the case.

As was mentioned earlier, storable propellant systems were emphasized in this study because of minimum development time. Since one cryogenic engine, the RL-10, is now available, consideration was given to this unit. It was not selected because the thrust level is too high and the design arrangement presented the landing gear and compactness problems discussed earlier.

Based on the considerations reviewed above, System No. 6, Table 6-V, defined as consisting of: (a) three engines, (b) fixed area injector, (c) gimbaled engines, in a cluster with a throttle ratio of 3.3 per engine was selected and is recommended for the LEM vehicle. A detailed description of the entire recommended propulsion system is provided in Paragraph 6.5 This selected system is essentially the same as the three-engine, fixed injector engine arrangement evaluated in Paragraph 6.3.2 except that the engine size was reduced to improve the vehicle arrangement. This size reduction was accomplished by increasing the chamber pressure from 100 psia to 150 psia and reducing the expansion ratio from 40:1 to 30:1.

6.4.2 Performance of Selected System

The final propulsion system selected for the LEM is exactly the same as the three-engine, variable injector system evaluated in Section 6.3.2, except for two changes. These consist of an engine size reduction, as discussed in the preceding Paragraph 6.4.1, and a weight improvement change in the crew station (reduction from 5187 lbs. to 4738 lbs.).

The performance of this system is defined in Figure 6-9. In this figure, the LEM initial weight is plotted as a function of thrust to weight ratio with altitude as a parameter. Altitude is a very significant factor in determining the thrust to weight ratio required for a selected trajectory. This is true because line-of-sight between the LEM and the selected lunar landing site is assumed to be required. To show the relationship between thrust to weight ratio and altitude a horizon line is superimposed on the altitude curves. As can be seen, if the line-of-sight requirement is met,

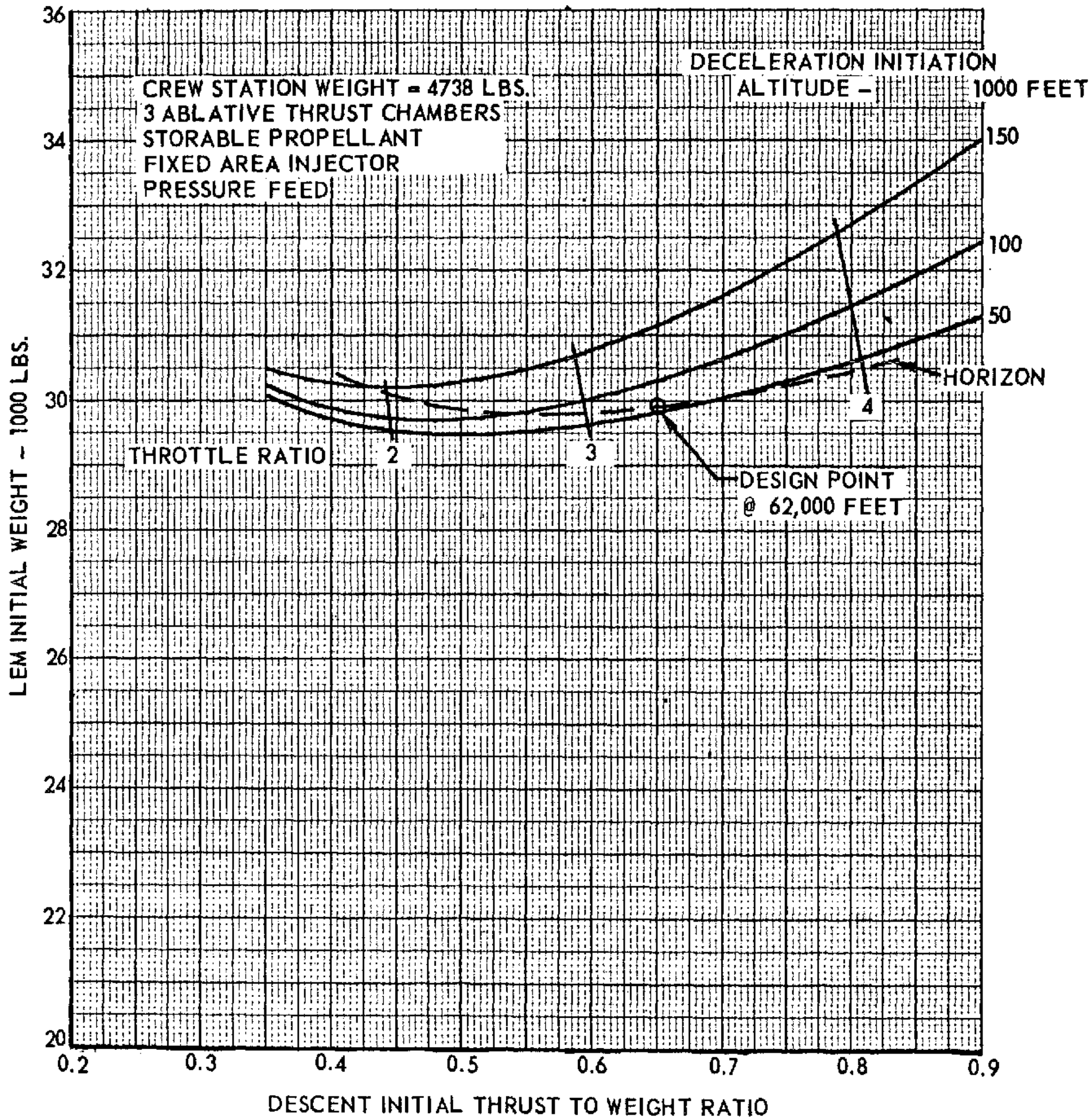


Figure 6–9 ENGINE THRUST REQUIREMENTS

the vehicle weight becomes almost independent of engine performance (thrust to weight ratio) over a wide range. In practice this means that significant changes can occur during the design of the basic vehicle without affecting the engine development program. In order to insure maximum growth potential for the LEM system, a thrust to weight ratio at the point of deceleration initiation was selected at 0.65 which corresponds to an altitude of 62,000 ft. With a vehicle weight of approximately 30,000 lbs., the thrust required is then 19,500 lbs. or 6500 lbs. per engine.

A sample calculation which illustrates the method used in obtaining the basic data presented in Figure 6-9 is shown in Appendix 6E, Paragraph 2.0.

6.4.3 Justification of Significant Systems

The most significant items in selecting the LEM propulsion system are: (a) propellants, (b) propellant feed systems, (c) vehicle staging, and (d) engine and related components. Each of these is discussed below and reasons for the choice made are given.

(a) Propellants

Several possible propellant combinations were considered for use in the LEM vehicle. As a result of these considerations, a discussion of which is presented in Appendix 6A, propellant combinations consisting of: (a) nitrogen tetroxide and Aerozine-50 and (b) liquid oxygen and liquid hydrogen, were selected for further evaluation. These evaluations, reported in Paragraph 6.2, show that either propellant combination provides adequate performance for the LEM vehicle. The storable propellants were selected over cryogenics. This was primarily because of the greater time requirement anticipated for the development of cryogenic systems. The nitrogen tetroxide and Aerozine-50 propellant combination was used in this study because it is the only high performance storable bi-propellant combination with significant development background. Furthermore, additional development effort will be applied to these propellants as work on the Titan III and Apollo Service Module engines continues.

(b) Propellant Feed Systems

With the selection of storable propellants for the LEM vehicle, the type of propellant feed system was clearly indicated to be a pressure feed design. The pump feed system is precluded because, up until the present time, no pump has been built which will operate with nitrogen tetroxide. The pressure feed system selected utilizes cold helium gas, stored at 3000 psia. This selection is based primarily on the simplicity and reliability of this type system. Several other methods were considered and rejected in this study. These are described in Appendix 6C. Positive expulsion of propellant was not provided nor considered necessary; since the vehicle will operate for the most part in a lunar gravity field. When operating in a zero "g" field, the reaction control system -- which utilizes positive expulsion -- is used for propellant "settling".

(c) Vehicle Staging

The general arrangement and size of the LEM vehicle is greatly affected by the staging concept selected. Propulsion system parametric studies, presented in Paragraph 6.2, along with system design studies indicate the compatibility of the vehicle with the Apollo system is compromised more with a two stage tandem arrangement than with a so-called "1-1/2-stage" (tanks and pressurant system staged) design. Furthermore, the 1-1/2-stage design tends to offer a more favorable landing arrangement, with better landing stability characteristics. An apparent advantage offered by the 2-stage vehicle is its inherent provision for protecting the ascent engine from possible lunar surface debris during the hovering and landing mode. More detailed knowledge of the lunar surface condition could dictate a two stage vehicle arrangement.

(d) Engine

The major factors considered in establishing the recommended engine system are: thrust vectoring, thrust misalignment, thrust modulation (throttling), number of engines and the method employed for achieving thrust chamber and nozzle cooling. The LEM initial thrust to weight ratio, i.e., 0.65, established by the lunar landing guidance requirements, fixes the required engine throttle ratio at approximately 10 to 1. This requirement is very significant in defining the thrust control system and in the selection of the thrust chamber cooling technique. The following paragraphs review the consideration used in establishing the engine system selection.

(1) Thrust Vector Control System

After careful review of the various techniques for effecting mechanical thrust vector control - including jet vanes, jetavators, secondary gas injection, gimbaling, etc. - gimbaling was selected as the only technique acceptable, from a state of the art standpoint, for the LEM application.

In the LEM design, vehicle control is achieved by both gimbal action and reaction controls. Such an arrangement offers the following advantages: provides for efficient and effective control, with maximum redundancy, and reduces the problem of thrust vector alignment associated with fixed engine installations. To insure reliability the gimbal system is operated by a tandem actuator which utilizes separate sources of hydraulic power. In the case of failure of one of the hydraulic systems, only one-half of the gimbal control authority is lost. Since the total control authority is shared equally between the gimbal system and the reaction control system - which is also redundant within itself - a failure of one hydraulic system results in a loss of only 25% of the total control. With such a loss the mission could still be completed. With the loss of another 25% of control authority, abort could still be effected.

In addition to providing basic vehicle control the gimbal system eliminates the problem of thrust vector misalignment common to fixed engine installations.

(2) Thrust Modulation (Throttling)

The LEM mission requires an engine throttle ratio capability of approximately 10 to 1. Current throttleable engines achieve a maximum operational state of the art in throttle ratio of approximately 3 to 1. These ratios are accomplished by the use of a propellant feed line throttle valve and a fixed area injector. The required 10 to 1 throttle ratio can easily be achieved with a variable area injector. However, variable area injectors are still in the laboratory stage of development and are not considered within the state of the art required for the LEM program. There are methods, however, which can utilize upstream line throttling to achieve the required throttle ratio. One obvious method is the use of upstream throttling and multiple engines. With this approach, three engines operated in the proper sequence can achieve the required 10 to 1 ratio. Other ramifications of this approach are feasible. For example, if multiple propellant lines are used to supply the injector, modulation plus step control of thrust can be achieved by operating upstream shut-off valves in conjunction with upstream throttle valves. Such an engine, however, has never been produced. For this reason, its development status is not consistent with the LEM program. The use of multiple engines and upstream throttling for achieving the throttle ratio required for the LOR mission has therefore been recommended.

(3) Thrust Chamber Cooling

Because of the limited cooling capacity of earth storable propellants, the throttle ratio of systems using regeneratively cooled chambers is limited to approximately 3 to 1. Also, in the pressure fed system, as selected for the LEM propulsion system, the pressure loss associated with regenerative cooling adds a significant weight penalty because of the increased tank pressure requirements. For these reasons as well as the apparent vulnerability of regenerative systems to micrometeoroid damage, the regenerative chamber was eliminated from consideration for the LEM system.

The use of ablative thrust chambers greatly simplifies the cooling problem over the regenerative system. While the ablative chamber is significantly heavier, it does not present the throttling nor pressure loss problems which are associated with regeneratively cooled chambers.

Ablative cooling systems have been used for some time in various applications. In the past few years significant steps have been accomplished in the development of ablators for thrust chamber and nozzle cooling. Currently, several such engines are under development. At least one ablative cooled engine in the 2000-lb. thrust class is nearing completion of its Pre-Flight Rating Test. Several other engine programs, such as the Apollo Service Module engine and the Gemini and Apollo Reaction Control motors, will significantly improve the development status of ablatively cooled engines.

On the basis of the above facts ablative thrust chambers have been recommended for the LEM propulsion system.

6.5 PROPULSION SYSTEM DESCRIPTION AND OPERATION

6.5.1 System Definition

The propulsion system selected and recommended for the LEM vehicle is defined below:

A. Propellants

Oxidizer	Nitrogen Tetroxide
Fuel	Aerozine 50*
Mixture Ratio	2:1
Amount**	21,166 lbs.
Descent Oxidizer	10,721 lbs.
Descent Fuel	5,361 lbs.
Ascent Oxidizer	3,389 lbs.
Ascent Fuel	1,695 lbs.

B. Engines

Number of Engines	3
Thrust Level	19,500 lbs. (6500 lbs/engine)
Propellant Feed	Pressure
Maximum Chamber Pressure	150 psia
Combustion Chamber Injector	Fixed
Throttle Control	Upstream Valve - 3.3:1
Nozzle Expansion Ratio	30 :1
Cooling	Ablative (800 seconds)
Vector Control	Gimbal
Maximum Motion	16°
Control Motion	±5°
Rate	30° per second
Acceleration	5 radians per (second)^2
Actuation	Hydraulic Tandem Actuators

* Aerozine 50 is a 50-50 mixture of Hydrazine and UDMH

** Amounts shown include 10% Descent reserve and 5% Ascent reserve. For explanation of reserve philosophy, see Volume VII, Paragraph 17.5.

C. Pressurization System

Pressurant	Helium
Descent quantity	47.6 lbs.
Ascent quantity	20.0 lbs.
Storage Pressure	3000 psia
Number of Tanks	4
Descent Fuel	1
Descent Oxidizer	1
Ascent Fuel	1
Ascent Oxidizer	1
Tank Material	Titanium

D. Propellant Storage System

Number of Tanks	8
Descent Fuel	2
Descent Oxidizer	2
Ascent Fuel	1
Ascent Oxidizer	1
Master Fuel	1
Master Oxidizer	1
Tank Design Pressure	215 psia
Tank Material	Titanium
Thermal Protection	
Descent Tanks	Gold Coating
Ascent Tanks	0.10 inch multi-layer insulation and low emissivity coating

6.5.2 System Operation

Figure 6-10 shows a schematic diagram of the propulsion system defined above. The function of each of the major components is provided in Table 6-VI. A detailed description of how the system operates both in the descent and ascent phases is provided in the following paragraphs.

6.5.2.1 Descent Phase

A. System Operation

(1) Helium tank squibs, in the descent fuel and oxidizer circuits, are opened.

(2) Inlet and outlet burst diaphragms are opened on the four descent tanks and both master tanks. The descent systems are now pressurized down to manual master tank outlet valves.

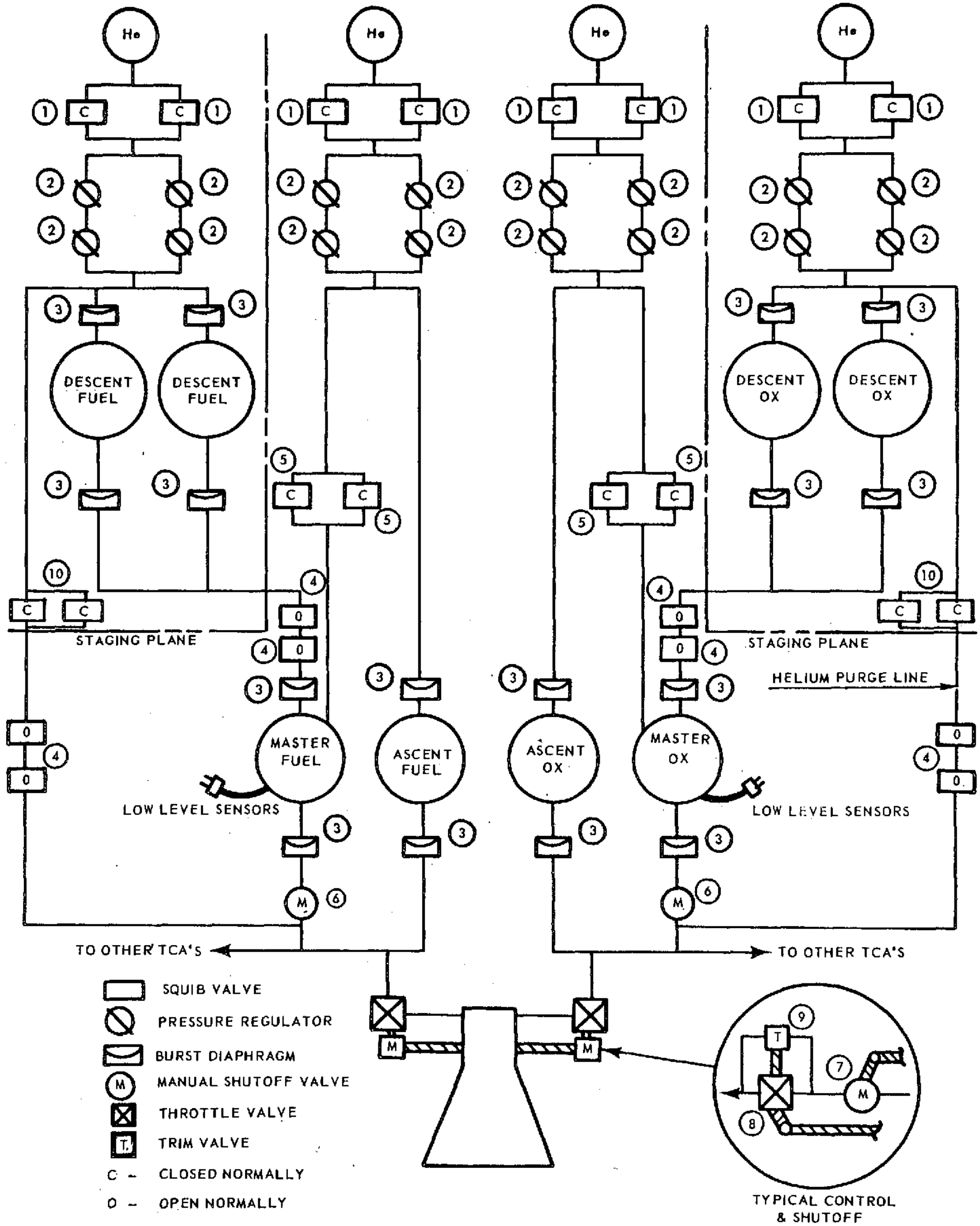


Figure 6-10 MAIN PROPULSION SYSTEM SCHEMATIC

TABLE 6-VI COMPONENT FUNCTIONAL DESCRIPTION

ITEM NO.*	COMPONENT	DESCRIPTION OF FUNCTIONS
1	Helium Tank Squib	Normally closed - serves only to contain helium within storage tank until system operation is initiated.
2	Helium Pressure Regulator	Reduces helium pressure from maximum tank pressure of 3000 psi to nominal operating pressures of 200 psia. Utilizes multi-stage regulator.
3	Propellant Tank Burst Diaphragm	Contains propellant within tank until initial usage required - reduces exposure of Item 2 to propellant - prevents propellant from accumulating in line during translunar zero "g" condition and freezing - reduces possibility of leakage and corrosion in feed lines.
4	Isolation Squibs	Normally open - driven closed prior to staging operation to isolate ascent tankage from descent system.
5	Ascent Pressurization Squibs	Normally closed - driven open to allow master tank pressurization for ascent - prevents descent operation from pressurizing launch tanks - prevents exposure of ascent helium pressure regulator to propellant during lunar residence.
6	Master Tank Shutoff*	Manually operated by mechanical linkage - seals off master tank on lunar surface to prevent propellant from entering engine feed lines and freezing - removes empty master tank from feed system during ascent to prevent helium disturbance of mixture ratio.
7	Engine Shutoffs*	Manually operated by mechanical linkage - shuts down all engines during coast - cuts out individual engine for throttling purposes - isolates malfunctioned engine from feed circuit.
8	Throttle Control*	Manually operated by mechanical linkage - provides thrust control by varying feed line pressure drop - control on all three engines "ganged".
9	Mixture Ratio Trim Valve	Automatic control with manual bias adjustment - maintains proper mixture ratio based on pressure and Item 8 - will not supercede Item 8 in throttling authority.
10	Purge Squibs	Normally closed - driven open after lunar descent to allow helium purging of engine feed lines.

*NOTES:
1. Manual shutoffs and throttles may also be operated by automatic control, but manual operation has prime authority.
2. Item number refers to Figure 6-10.

(3) The master tank shut-off valves are opened and throttles are set at the desired power. The manual* engine shut-off valves are then opened to initiate engine firing to accomplish retro to transfer orbit.

(4) Manual* shut-off valves at the engines are closed to stop firing during coast. The throttle control valves are left in the open position in preparation for the next firing.

(5) Manual shut-off valves are opened at the engines to initiate thrust for final descent and hover.

(6) Individual engines are shut down as required prior to landing by closing the manual engine valve. Throttle valves are operated as a "gang" so that non-operating engines are immediately available at the proper power setting in an emergency.

(7) After landing, preparation for lunar stay is effected by closing master tank valves and opening manual engine valves on all engines. Purging is accomplished by opening engine throttles and firing helium purge squibs. After purging is complete, the purge lines are closed by firing squibs. Throttles are left open in preparation for launch.

(8) Squib valves between the ascent and descent tanks are fired and separation of the system lines is effected at the staging plane.

B. Propellant Management

Propellant feeds from the four descent tanks through the two master tanks and on to the thrust chamber. At the point where all the normally required descent propellant has been consumed, the descent tanks are empty and, for a normal mission, lunar landing has been accomplished. However, should any descent reserve propellant be required, it would be drawn from the master tank with pressurization provided by the descent system through the then empty descent tanks. At the completion of the descent maneuver, all unused descent reserve propellant is automatically available for launch by virtue of the fact that it was initially loaded in the master tanks. The feed lines from the master tanks to the engine are then purged with descent helium and the descent tanks and pressurization systems are staged.

C. Engine Operation

The three engines are fired at full thrust (19, 500 lbs) to initiate deceleration. They are then throttled equally until the trajectory thrust requirements decrease to 13, 000 lbs. At this time, one engine is cut off and the other two engines are increased to full thrust. The two burning engines are then throttled equally to provide trajectory thrust requirements until the end of deceleration. At the initiation of hover, another engine is cut off leaving one engine to provide hover thrust. The system operating time for each phase of descent is shown as follows:

---

*NOTE: Manual valves and throttles may also be operated by automatic control but manual operation has prime authority.

Phase	Operating Time - Seconds
Retro	26
Deceleration	333
Hover	60
Translation and Let down	45

It should be noted that there is no requirement for engine restart during descent except in case of an engine failure during hover. During hover, the engine feed lines will be filled with propellant; hence, another engine can be restarted immediately. The time required for restart is the pilot response time plus approximately 100 milliseconds maximum .

6.5.2.2 Ascent Phase

A. System Operation

(1) Helium tank squibs in the ascent fuel and oxidizer circuits are opened. Inlet and outlet diaphragms are then opened on the ascent tanks.

(2) The squib valves and manual valves on master fuel and oxidizer tanks are opened. The system is now pressurized down to manual valves on each tank.

(3) The engine shutoff valves are opened. The engines are shut off selectively after clearing the landing gear.

(4) The master tank manual valves are closed during ascent when low level indicator shows master tanks are expended.

(5) The remaining operations are accomplished in a similar manner to the landing operation, using engines selectively as needed.

NOTE: Bypass trimmer valves sense fuel and oxidizer flow and maintain thrust and mixture ratio constant at any power setting. They may be biased differently or closed manually if necessary.

B. Propellant Management

Propellant for ascent is furnished from the two ascent tanks and the two master tanks directly to the engines. Pressurization for this maneuver is derived from a separate ascent system. Feed lines will be built such that the difference in pressure drop between the ascent tank lines and those of the master tanks will insure that the latter become empty first. At this point, the master tanks are manually shut off and the remainder of the required propellant is drawn from the ascent tanks.

C. Engine Operation

For ascent, only one of the three engines is required. The engine with minimum descent operating time is used, if visual inspection indicates it is in operating condition. If this is not the case, either of the other engines may be used, since each has adequate operating time remaining. In any case, the other engines provide redundancy in ascent. The system operating time for each phase of ascent is shown below:

Phase	Operating Time-Seconds
Boost	228
Plane Change	1

6.6 DEVELOPMENT

The purpose of this section is to define the proposed propulsion system development schedule and to outline possible development problem areas. The proposed approach is to divide the system development into two parallel programs: (1) the engine development program, and (2) the propellant storage and pressurization system development program. The interface between these two systems is the engine inlet upstream of the engine control valves. The engine package, consisting of three thrust chambers, the thrust chamber mounts, and the chamber control valves should be developed by an engine manufacturer. The propellant tanks, lines, and pressurization system are dependent to a large extent on vehicle configuration and installation requirements. Therefore, the responsibility for this system should be retained by the prime contractor. With this division of responsibility, tanks and propellant system components would be a prime contractor function. Propellant system valves, rupture diaphragms, sensing devices, and pyrotechnic components would be developed by component manufacturers from design requirements determined by the prime contractor. This division of responsibility relieves the engine manufacturer of detailed component responsibility and provides an opportunity for better coordination and control of the over-all system by the responsible prime contractor.

The proposed development schedule is presented in Figure 6-11. This schedule requires delivery of the first qualified engine 21 months after program go-ahead. Accomplishment of this objective requires maximum use of existing components and use of current state of the art design practices in those instances where a new component must be developed.

The major engine development problem areas which are anticipated are: (1) the ablator, (2) the engine throttling control valves, and (3) the engine injector. Ablative thrust chambers have been designed, built and operated but not at the proposed thrust level and time duration and not for throttleable engines. Heat flux determinations, erosion characteristics, and char depth must be determined from hot firing tests. The major injector problems will be concerned with elimination of flame fans and engine hot spots.

The propellant storage and pressurization system development should be conducted parallel to the engine development. The specific propellant problems related to the LEM configuration are: (1) foaming in the tanks resulting from the series arrangement of tanks, feed line vortexing, and introduction of pressurant; (2) propellant splash and swirl due to tank motion; (3) propellant metering and utilization. Considerable component and systems testing will be required to solve these problems and to demonstrate the ability of the system to deliver the propellant to the engine at the conditions required.

Ground testing of the complete propulsion system should include cold flow tests, vibration tests, environmental tests, propellant loading and ground handling tests, and complete system hot firing tests. Flight testing should include both tethered flight and earth orbital testing. In these tests, response times, pilot functions, propellant utilization, zero 'g' operation, and space environment operation should be evaluated.

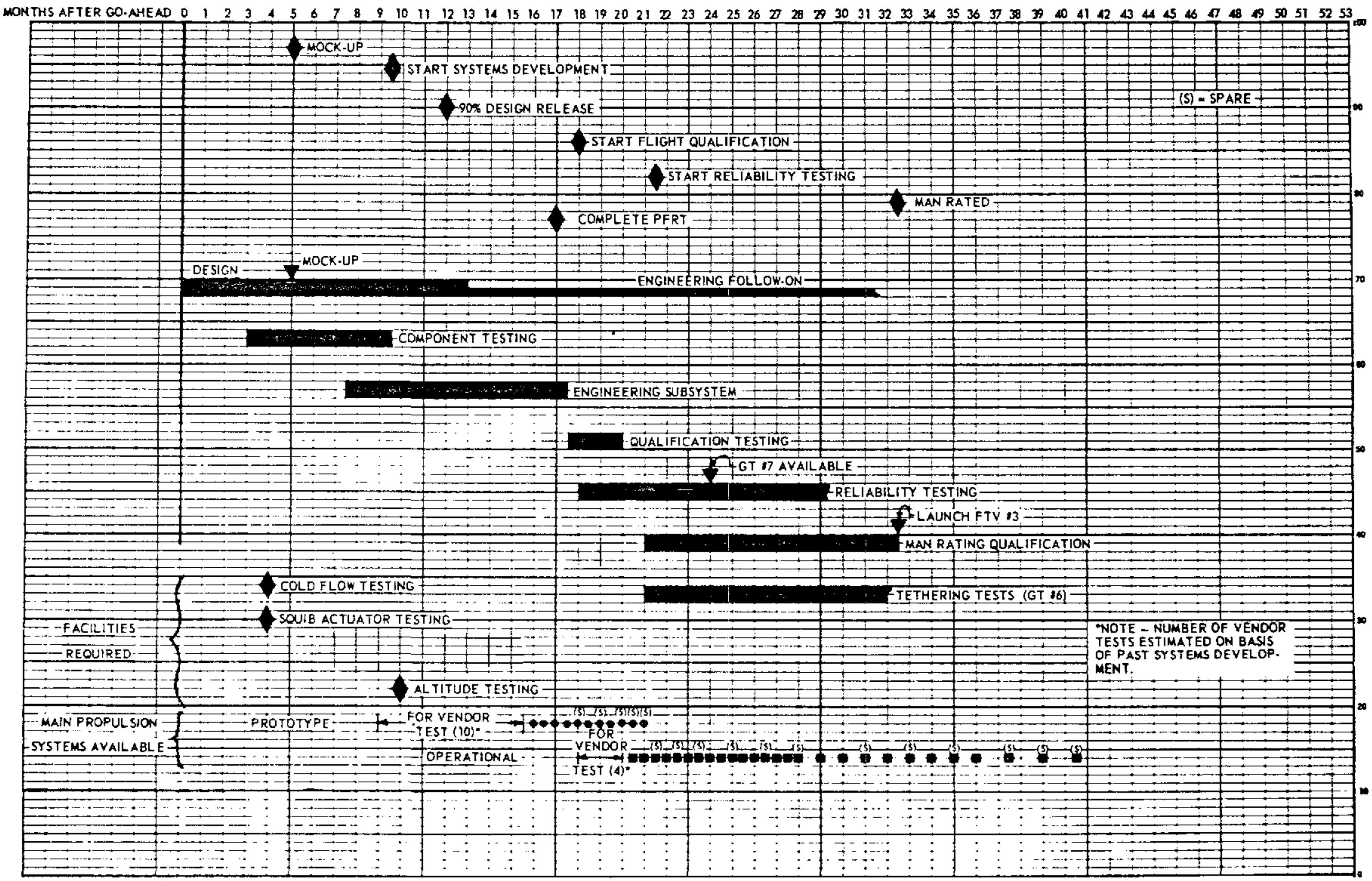


Figure 6–11 PROPULSION DEVELOPMENT SCHEDULE

Development and qualification of a suitable propulsion system represents one of the major problems in the overall LEM development. Therefore, detailed attention has been given during this study to establishing a realistic schedule. The proposed schedule has been discussed with the major engine contractors and the concensus is that the proposed program is realistic and can be accomplished successfully.

REACTION CONTROL

SYSTEM

7.0 REACTION CONTROL SYSTEM

7.1 INTRODUCTION

This section presents the results of the reaction control system studies which were conducted as a part of the LEM study program. The objective of the reaction control portion of the study was to select and provide a preliminary design of the most feasible reaction control system which would meet the LEM requirements.

The initial effort was directed to a study of the LEM mission to determine the functions which could best be accomplished by the reaction control system (RCS). Requirements for the system were defined and analyses were accomplished to aid in the selection of a system concept. The selected system was then defined in detail and the development potential was examined. The selected system meets the operational requirements of the LEM and can be developed within the required time period.

7.2 SUMMARY

The Reaction Control System (RCS) was designed to perform the following functions: (1) separation, (2) rendezvous, (3) docking, and (4) vehicle stability and control. The system utilizes a total of 16 motors of the size, quantity and functions shown below:

Function	No.	Thrust per Engine
Roll	4	50 lbs.
Yaw	4	100 lbs.
Pitch	4	100 lbs.
Translation for rendezvous	4	200 lbs.

Separation from the Command Module is accomplished by use of the pitch and yaw motors. Rendezvous is effected by using the translation motor, in conjunction with the other attitude controls. Docking employs the normal attitude control system with the pitch and yaw motors furnishing the final docking motion. To achieve vehicle stability and control, the RCS operates in conjunction with an engine gimbal system. The control authority is shared equally between the systems.

The system utilizes N_2O_4 and Aerozine-50 as propellants. The propellant feed system is a positive expulsion pressure type, which uses cold helium gas as the pressurant. Control of thrust is achieved by pulse width modulation. The thrust chambers are ablative and completey redundant.

It is expected that no problem will be encountered in developing the RCS motors in time for the LEM program. Extensive development on the Apollo and Gemini 50-lb. and 100-lb. thrust RCS motors will aid this program greatly.

7.3 MISSION REQUIREMENTS

Examination of the LEM mission shows that in addition to the stability and control functions normally assigned to the RCS, other propulsion requirements can logically be accomplished by the RCS in preference to the main propulsion system. Therefore, the RCS requirements of the LEM include both propulsion and vehicle control. These requirements are discussed separately in the following paragraphs.

7.3.1 Propulsion

The vehicle propulsion maneuvers which are supplied by the RCS are:

(a) Separation $\Delta V = 10$ feet per second
(b) Rendezvous $\Delta V = 200$ feet per second
(c) Docking $\Delta V = 10$ feet per second

The separation maneuver requires a small amount of energy and would be difficult to accomplish accurately with the large engines of the main propulsion system. The RCS must be operative for the separation maneuver. Use of the system for separation eliminates the necessity for a start of the main propulsion system.

The rendezvous maneuver is characterized by many small impulse intervals followed by slight corrections which are commanded from line-of-sight observations. If the main propulsion system is used for this operation, the vehicle must be rotated for each correction to align the thrust axis in the required impulse direction. This type maneuvering degrades the accuracy of the line-of-sight observations and requires considerable reaction control propellant to position the vehicle accurately. Therefore, the capability of translating fore and aft along the pilot's normal line-of-sight is extremely desirable. The thrust level which allows this maneuver to be accomplished most efficiently and accurately is 400 pounds (see Volume I, Section 7.0). Use of the RCS to perform this maneuver greatly enhances its success.

The docking maneuver is a primary function of the command and service module. However, propellant allowances for the docking maneuver have been included in the LEM reaction control to provide redundancy for this critical operation.

Zero "g" and/or negative "g" operation of the RCS necessitates employment of positive propellant expulsion for the LEM system. Since this requirement exists, it follows that use of the RCS for propellant settling eliminates positive expulsion requirements on the main engine system. This has also been established as an RCS requirement.

7.3.2 Stabilization and Control

The RCS provides only roll control during propulsion system operation. The main engines are gimbaled to provide pitch and yaw control. During coast operations the RCS provides pitch, yaw, and roll control. Detailed development of the system requirements for the vehicle is presented in Volume IV, Section 13. A summary is presented below.

The torque levels produced by the RCS are defined by consideration of the maximum angular acceleration required at maximum moment of inertia. The maximum required torques were averaged and adjusted in each axis in an attempt to achieve symmetry and reduce the number of engine sizes. The maximum torques required were found to be 2000 ft-lbs. for pitch and yaw control and 1000 ft-lbs. for roll control.

The above conditions establish the maximum required torques for vehicle control. Lesser average torques can be obtained by "pulse-width modulation" of the reaction control engines. Short minimum pulse-widths are desired for the following reasons:

(a) Minimize propellant used during limit cycle operation.
(b) Keep pulse-width modulation period short compared to the natural period of the vehicle stabilization system.

A minimum pulse-width of 0.020 seconds was selected as being representative of the state of the art of solenoid propellant valves. A shorter minimum pulse-width would be desirable to conserve propellant but satisfactory stabilization is obtained with a pulse-width of 0.020 seconds.

The stabilization system characteristics and the effect of vehicle C.G. travel were examined relative to the necessary trajectory control and maneuvering to establish the total impulse requirements. Based on this, the duty cycles were established in terms of per cent flight time for each phase of the mission. These data are presented in Table 7-I.

TABLE 7-I
PROPELLANT WEIGHT REQUIREMENTS

A. PROPULSION FUNCTIONS

MANEUVER	VELOCITY CHANGE FT/SEC	INITIAL VEHICLE WEIGHT POUNDS	*PROPELLANT CONSUMED – POUNDS
SEPARATION	10	29,887	31
RENDEZVOUS	200	6,083	127
DOCKING	10	5,956	6

TOTAL 164

B. CONTROL FUNCTIONS

CONTROL	TIME – SECONDS	CHAMBERS REQUIRED	THRUST PER CHAMBER – POUNDS	DUTY CYCLE – %	*PROPELLANT CONSUMED – POUNDS
A. DESCENT					
1. MAIN ENGINE BURNING:					
ROLL	464	4	50	0.5	1.54
2. COASTING:					
ROLL	1740	4	50	0.5	5.8
PITCH	1740	4	100	2.0	46.45
YAW	1740	4	100	1.2	27.8
B. ASCENT					
1. MAIN ENGINE BURNING:					
ROLL	243	4	50	0.5	.81
2. COASTING:					
ROLL	3420	4	50	0.25	5.7
PITCH	3420	4	100	0.75	34.2
YAW	3420	4	100	0.5	22.8

TOTAL 145.1

*SPECIFIC IMPULSE – 300 SECS.

7.4 SYSTEM ANALYSIS

7.4.1 Concept Selection

The following state-of-the-art propellants were reviewed in determining the recommended RCS for the LEM:

(a) Nitrogen (cold stored gas)
(b) Freon (heated stored gas)
(c) Hydrogen peroxide (monopropellant)
(d) Nitrogen tetroxide - Aerozine-50 (bipropellant)

The evaluation of these propellants resulted in the following conclusions:

(a) Nitrogen expelled from high pressure storage tanks provides the lowest performancebut is the most reliable and simplest approach. However, a nitrogen system is excessively heavy. Even if the propulsion operations were removed from the RCS requirements, a nitrogen system would still weigh approximately 4 times as much as the selected bipropellant system.

(b) Freon exhibits better performance than nitrogen, but the process of heating and expelling the freon requires a complex system which would be difficult to control during an LEM type mission.

(c) Hydrogen peroxide exhibits intermediate performance, and has been used in space on manned vehicles for short mission times. However, long term storage is a problem because of deterioration of the propellant by catalytic reaction with most contaminants. The advantage with this approach in assuring mission success is questionable when compared to the higher performance bipropellants.

(d) Nitrogen tetroxide - Aerozine-50 exhibit excellent performance and controllability. While the system is relatively complex it contains no basic problems which have not previously been solved in propulsion systems development. Development of reaction control systems with these propellants is underway for Gemini and Apollo.

In view of the above conclusions, a nitrogen tetroxide-Aerozine-50 bipropellant system was selected for the LEM. The system employs positive expulsion pressure feed to ablative thrust chambers. The feed pressure is supplied by regulated high pressure stored helium gas. Chamber operation is controlled by solenoid operated propellant valves attached to each chamber.

7.4.2 System Sizing

Sizing of the reaction control system must be accomplished in two steps. Since the control requirement is in terms of torque, the chamber positions must be established before their size and the quantity of propellant can be determined. The most desirable condition is to provide the maximum moment arm between each chamber and the vehicle C.G. This results in the smallest chambers and minimum propellant expenditures.

A review of the vehicle configuration showed that a practical maximum arm length is 10 feet (from the vehicle roll axis). The resultant location and size of the yaw, pitch, and roll chambers are shown by Figure 7-1. The thrust values shown are the minimum thrusts required to satisfy the maximum torque requirements presented in Paragraph 7.3.2 of this volume.

The translation thrust requirement given in Paragraph 7.3.1 is satisfied by the 200 pound thrust chambers located as shown in Figure 7-1. These chambers are located with 2.5 foot moment arms so that the roll chambers can counteract the torque produced if one of the translation chambers should fail to fire. The total number of chambers is 16, consisting of: 4 yaw, 4 pitch, 4 roll, and 4 translation.

The required propellant can be determined once the chamber sizes and number are established. A propellant specific impulse of 300 seconds was used. This was selected rather than the value of 320 seconds which is attainable with these propellants because of the pulse modulation. The minimum pulse-width of 0.02 seconds will cause a decrease in net specific impulse because of the large number of stops and starts relative to the total burning time.

The procedure for calculating the propellant required is necessarily different for the propulsion and the control modes of operation. The following methods were used:

(a) Propulsion - Propellant consumed weight was calculated for each of the mission propulsion requirements, based on the trajectory characteristic velocity equation:

$$\mu = \frac{m_o}{m_f} = e^{(\Delta V / I_{sp}\, g)}$$

m_o = initial weight

m_f = final weight

ΔV = mission characteristic velocity change

I_{sp} = specific impulse - sec.

g = 32.174 ft/sec^2

The propellant consumed is the difference between the initial and final vehicle weight for each maneuver.

(b) Control - Propellant consumed for vehicle stabilization and control is obtained by multiplying the thrust of the motors being operated by their duty cycle time and dividing the result by the motor specific impulse.

The inputs and results of the propellant quantity calculations are shown in Table 7-I. The total propellant required for the mission is 309.1 lbs. In order to provide a margin for control error, unavailable propellants, and main propulsion system propellant settling, a reserve of 12% was added. Thus, the total reaction control propellant will be 346.1 lbs. The best

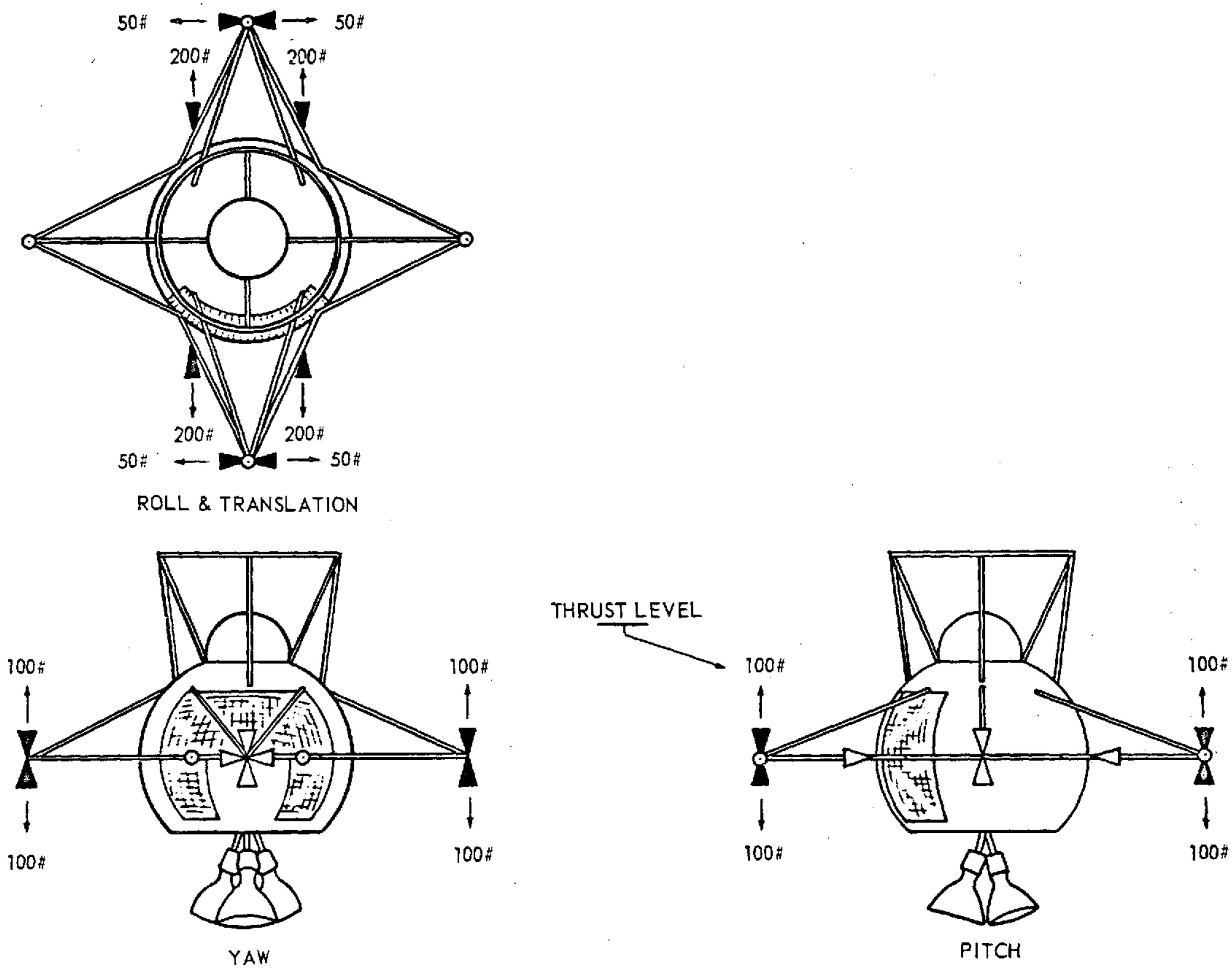


Figure 7-1 REACTION CONTROL THRUST CHAMBER LOCATION

performance oxidizer to fuel mixture ratio of the propellants is 2. Using this ratio, the tanks are sized for 115.4 lbs. of Aerozine-50 and 230.7 lbs. of nitrogen tetroxide.

The four 100 pound thrust engines, firing aft provide more than 0.01 "g" at LEM maximum gross weight. Considering that the tanks are almost full for the zero "g" start times of the mission, this "g" field is adequate for settling of the main propellant tanks. Burning time for settling was estimated to be less than five seconds with negligible propellant consumption for this operation.

7.5 SYSTEM DEFINITION

7.5.1 Physical Description

The selected RCS is a pulse-modulated pressure-fed positive expulsion system which distributes storable propellants (nitrogen tetroxide - Aerozine-50) to 16 ablative cooled thrust chambers. Four thrust chambers, coupled in pairs, provide stabilization and torques around each of the three major axes as shown in Figure 7-1. The pitch and yaw chambers are aligned parallel to the vertical axis of the vehicle. Translation along the vertical axis can be obtained by firing the pitch and yaw motors in pairs to move in either direction. This mode of operation provides the separation and docking maneuvers as well as providing propellant tank settling for the main propulsion system. An additional four thrust chambers fire parallel to the pilot's normal line-of-sight to provide translation propulsion for rendezvous.

A. Feed System

A schematic of the RCS is shown on Figure 7-2. Pressure feed is provided by 0.6 lbs. of helium stored at 3000 psia in a spherical titanium tank. The pressurized helium is discharged through redundant pressure regulators. The propellant tanks provide positive propellant expulsion for zero and negative "g" operation through the use of bladders which are designed for multiple cycling. The propellant tanks are mounted to the crew station structure and are insulated to prevent propellant freezing (Volume III, Appendix 6C). The propellant tanks are sized to contain 115.4 lbs. of Aerozine-50 and 230.7 lbs. of nitrogen tetroxide.

The propellant is routed to the thrust chambers through feed lines and redundant dynamic components. The feed lines are insulated and electrically heated to prevent propellant freezing.

Manual shut-down valves are provided at the exits of the pressurization and propellant tanks for long inert lunar stay periods as a positive means of preventing leakage.

B. Thrust Chambers

Thrust chambers are ablation cooled and have an expansion ratio of 40:1. The operating chamber pressure is 150 psia. The chambers provide the thrust levels shown by Figure 7-1. The chambers are capable of operating in either a pulsing or continuous mode for 15 minutes.

As shown by Figure 7-1, the pitch, roll, and yaw chambers are mounted on 10 foot arms located approximately in a plane common with the vehicle center of gravity. The rendezvous chambers are mounted on the crew station structure in the same plane at a distance of 2.5 feet from the vehicle center of gravity.

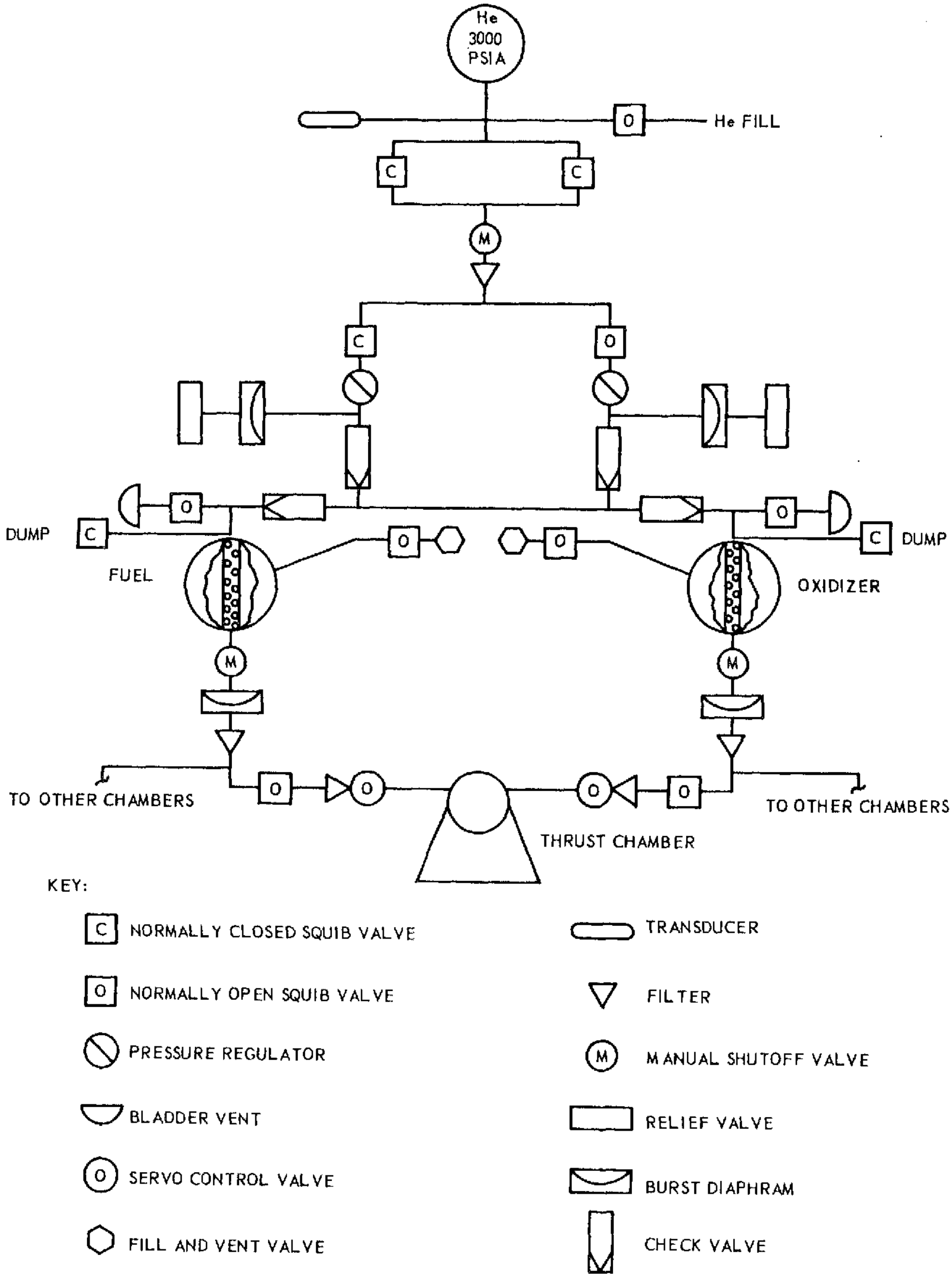


Figure 7-2 REACTION CONTROL SYSTEM SCHEMATIC

Control of the chambers is provided by two solenoid operated valves mounted on each chamber. The valves are capable of providing a minimum pulse-width of 0.020 seconds.

7.5.2 Operational Description

A. System

The RCS is actuated just prior to LEM separation by opening the normally closed squib actuated valves on the pressurization system (see Figure 7-2). Helium then flows through the filter and normally open squib valve, through the primary regulator, to the propellant tanks to provide feed system pressurization. The check valves downstream of the regulators provide isolation of the regulator - relief valve systems in the event of a primary regulator malfunction.

The burst diaphragms are opened to permit propellant flow to the solenoid control valves. These diaphragms provide positive propellant tank downstream isolation prior to burst. It should be noted that propellant tank upstream isolation is provided by the positive expulsion bladders.

The thrust chambers are individually controlled on command by signals to the solenoid valves. The pair of solenoid valves on each chamber are mechanically connected to provide positive propellant flow control.

During long lunar stay periods, the pressurization system manual shut-off valve is closed to prevent leakage, and the propellant feed line manual valves are closed to safeguard against loss of propellant.

B. Mission

The RCS provides stabilization and torques around the vehicle major axes and translation propulsion for separation, docking, and rendezvous. The mission functions of the reaction control system are discussed in the following paragraphs.

<u>Separation</u> - During separation, attitude control is provided in all axes. In addition, translation is provided by selective usage of the pitch and yaw engines.

<u>Orbit Transfer</u> - Vehicle attitude control is provided prior to main propulsion system firing and during the coast phase. The pitch and yaw engines provide main propulsion system propellant tank settling. Roll control is provided at all times.

<u>Descent</u> - Propellant tank settling is provided prior to main propulsion system firing by the pitch and yaw engines. Roll control is provided at all times during descent.

Hover and Translation - Roll control is provided during hover and translation.

Lunar Boost - Roll control is provided during lunar boost.

Plane Change - The pitch and yaw engines provide main propellant tank settling prior to main propulsion system firing. Roll control is provided at all times during plane change.

Rendezvous - Rendezvous propulsion is provided by the translation engines. In addition, attitude control is provided at all times.

Docking - Attitude control is provided during the docking maneuver. In addition, selective usage of the pitch and yaw engines provides the translation necessary to accomplish docking.

7.5.3 Redundancy

A. Component Redundany

System overall reliability is based on the concept that tanks, feed lines, and other static components have an extremely high inherent reliability whereas valves, regulators, and other dynamic components are less reliable. In determining redundancy requirements, the criteria was that dual failures of dynamic components must occur before a mission abort is required. Redundant flow paths are provided in the pressurization system. Redundant solenoid control valves are not provided since a single valve failure will necessitate shutdown of only one chamber while the remaining engine in the couple provides the necessary control torque.

B. Control System Redundancy

Pitch and Yaw - The pitch and yaw engines provide redundancy to the main engine gimbal system during main engine thrusting periods. In addition, these engines provide pitch and yaw redundancy since one engine in a couple will provide satisfactory control torques. Operation of one coupled engine will not produce a pure couple since a translation vector will be introduced. This vector can be nullified by firing the two engines in the other axis since all pitch and yaw engines have their thrust axis parallel to the vehicle main engine thrust axis.

Roll - Roll redundancy is provided by the translation engines since they are located such that they will provide the same control torques as the roll engines.

7.5.4 Weight

Component weights were obtained from manufacturers' estimates and existing component weights which were generated for the Apollo study effort. The required propellant weights were calculated as discussed in Paragraph 7.4.2. A weights summary for the selected RCS is presented in Table 7-II.

Table 7-II REACTION CONTROL SYSTEM WEIGHT

A. HARDWARE			
COMPONENT	INDIVIDUAL (POUNDS)	NUMBER USED	TOTAL (POUNDS)
THRUST CHAMBERS			
50# THRUST	3.7	4	14.8
100# THRUST	5.75	8	46.0
200# THRUST	7.5	4	30.0
(VALVES INCLUDED)			
TANKAGE			
OXIDIZER	6.9	1	6.8
FUEL	4.54	1	5.9
PRESSURANT	7.66	1	7.7
MOUNTS & INSULATION INCL.			
SQUIB VALVES	0.16	43	6.2
RELIEF VALVES	.5	2	2.0
FILTERS	1.0	2	2.0
PRESSURE REGULATORS	1.0	2	2.0
BURST DIAPHRAMS	.5	4	2.0
CHECK VALVES	.5	4	2.0
LINES & FITTINGS			14.5
CIRCUITRY			4.9
TOTAL			146.8
B. PROPELLANT			
PROPULSION			164.0
CONTROL			145.1
RESERVE (12%)			37.0
TOTAL			346.1
C. PRESSURANT			
HELIUM			.48
RESERVE (25%)			.12
TOTAL			.6
TOTAL SYSTEM LOADED WEIGHT			493.5

7.6 DEVELOPMENT

The consensus of RCS manufacturers is that development of the proposed RCS to man rated status can be accomplished in 32 months. Man-rating is not considered to have been established until the system has experienced the first three near earth space flights. The detail program plan is presented in Volume VIII, Section 1.2. Qualification for man-rating is assured by reliability testing of the system at the manufacturer's facility, by testing in space environment simulation facilities, and by system evaluation in early earth orbital flights.

Preliminary qualification which will allow delivery of the system for vehicle integration and testing will be completed in 15 months. Development work presently accomplished and now in progress on the other manned space systems will provide the necessary lead information to enable this rapid development.

The major development effort will be required for the chambers and the positive expulsion tanks. Ablative chambers of 50 and 100 pounds thrust are now under development for Gemini and Apollo. Some experimental work has been done on 200 pound thrust chambers. However, the larger chambers are considered to be less difficult to develop and no problems of a serious nature should be encountered.

The positive expulsion provisions for the propellant tankage will require major development effort. The problem is made difficult by the long storage times and intermittent system operation. Positive expulsion has been successfully accomplished with earth storable propellants of this type in propulsion systems for space application. Bladders made from multilayer teflon have proved satisfactory. Recent successes have been experienced in the development and testing of folding metal diaphragms. Further study will be required to determine which of these concepts should be pursued in the LEM program. However, it is felt that either offers the potential for a successful system.

APPENDICES

APPENDIX 6A
PROPELLANT SELECTION

1. GENERAL

The objective of this portion of the study was to evaluate possible propellant combinations which could be employed in the LEM propulsion system. Many propellant combinations were considered in the initial evaluation, however, most were discarded for obvious reasons, such as lack of development, low performance, etc. The propellants which were considered in the final selection are presented in Table 6A-1. These propellants were rated with respect to the factors listed. These ratings are qualitative only, but are considered adequate for propellant selection. The preferred selection for each factor is indicated by the shaded blocks. The propellant evaluation was conducted independent of feed system design, engine configuration, vehicle configuration, or trajectory analysis. The data upon which the propellant evaluation chart was based were consolidated from various government publications and information supplied by major engine manufacturers.

2. PROPELLANTS CONSIDERED

To simplify the consideration of the possible propellant combinations, the propellants were divided into two general categories as follows:

(a) Earth storable propellants, which for this study were defined as those propellants which may be stored for extended periods at earth ambient environmental conditions in conventional materials.

(b) Cryogenic propellants, which must be stored at very low temperatures and require insulation or cooling or both for long time earth storage.

In the case of earth storable propellants, the oxidizers were predominant in establishing the rating. The hydrazine derivative fuels are very similar in most respects.

Nitrogen tetroxide and Aerozine-50 were selected as the most desirable storable propellant combination. These were chosen for this study because they are the only high performance storable bi-propellants with significant development background. Also, additional development effort is being applied to these propellants in conjunction with the Titan III and the Apollo Service Module engine programs. Some characteristics of these propellants, such as freezing temperatures, are not particularly desirable for this application. There are other possible storable propellant combinations, such as mixed oxides of nitrogen and monomethyl-hydrazine, which provide performance characteristics equal to the selected propellants. Furthermore, it appears that this particular combination may have better temperature characteristics. Although nitrogen tetroxide and Aerozine-50 were selected for study purposes, it appears that other storable propellants which exhibit equal or superior performance can be utilized in the resulting system.

Liquid hydrogen and liquid oxygen were also selected for concept evaluation because of their high performance, and because an engine is currently available which might be modified to accomplish the LEM mission.

TABLE 6A-I. PROPELLANT SELECTION

CRITERIA	EARTH STORABLES			CRYOGENIC			
	IRFNA UDMH	N_2O_4 AEROZINE - 50	CLF_3 MMH	OF_2 MMH	LO_2 LH_2	LO_2 RP-1	LF_2 LH_2
I_{sp} (VAC)	295	318	314	384	430	285	445
DENSITY IMPULSE	386	387	436	478	146	289	206
STATE OF ART	GOOD	GOOD	FAIR	POOR	GOOD	GOOD	POOR
STORABILITY	GOOD	GOOD	GOOD	FAIR	POOR	FAIR	POOR
HANDLING	GOOD	GOOD	FAIR	POOR	FAIR	GOOD	FAIR
STARTING	GOOD	GOOD	GOOD	GOOD	FAIR	FAIR	GOOD
MATERIALS	FAIR	GOOD	FAIR	POOR	FAIR	GOOD	FAIR
TOXICITY	FAIR	FAIR	FAIR	POOR	GOOD	GOOD	POOR

*A 50% MIXTURE OF HYDRAZINE AND UNSYMETRICAL DIMETHYL HYDRAZINE

3. EVALUATION CRITERIA

The criteria, used in Table 6A-I for evaluation of the propellants are defined below:

(a) I_{sp} (VAC) is the specific impulse which can be attained in a vacuum with the propellant combinations concerned. In most instances, the I_{sp} data presented are based on actual engine tests.

(b) Density Impulse is the propellant bulk density multiplied by the propellant vacuum specific impulse. It is inversely proportional to propellant system volume.

(c) State of the art refers the current development status of a component, item or system.

(d) Storability is a characteristic which defines the ability of a propellant to withstand the ambient conditions under which it is stored.

(e) Handling refers to the storage and transportation of the propellants from one place to another. Handling characteristics are affected by liquifaction temperature, toxocity, specific gravity, corrosive properties, etc. In the case of earth storable propellants, the major handling problems are associated with the oxidizer. With cryogenic fluids the fuel, liquid hydrogen, presents the greatest problem.

(f) Starting criteria include basic ignition, system delay time and system complexity, with hypergolic mixtures representing the most desirable characteristics.

(g) Materials compatibility refers to the ability of the propellant to be placed in contact with materials without a detrimental effect on the propellant and/or material. This characteristic was judged relative to the state of the art and availability of valves, seals and other components, including materials for storage containers.

(h) Toxicity is a measure of danger associated with breathing the propellant vapors. Both the toxicity of the propellant and the exhaust products were considered relative to ground service and flight problems.

4. PROPELLANT CHARACTERISTICS

Physical and thermodynamic characteristics of the propellants which were selected for study are shown in Tables 6A-II and 6A-III for standard temperature. These data and data on the effect of temperature on vapor pressure, density, heat capacity, and viscosity were obtained from periodicals, industry data, and other publications.

TABLE 6A-II

PHYSICAL AND THERMODYNAMIC PROPERTIES OF *AEROZINE-50 AND NITROGEN TETRAOXIDE

	Aerozine-50	Nitrogen Tetraoxide
Freezing Point (^oF) at 14.7 psia	+18.8	+11.8
Boiling Point (^oF) at 14.7 psia	158.2	70.1
Density of Liquid at 77^oF. (lb/ft^3)	56.1	89.52
Viscosity of Liquid at 77^oF. (lb/ft/sec)	54.9×10^{-5}	26.9×10^{-5}
Vapor Pressure at 77^oF. (psia)	2.75	17.7
Critical Temperature, calc. ^oF	634	316.8
Critical Pressure, calc. (psia)	1696	1469
Heat of Vaporization, calc. (BTU/lb)	425.8	178.1
Heat of Formation at 77^oF., calc. (BTU/Lb)	507.35	45.13
Specific Heat at 77^oF., calc. (BTU/Lb)	0.694	0.271
Thermal Conductivity at 77^oF., calc. (BTU/Ft/Hr/^{o})	0.151	0.0717

*Aerozine-50 is a 50-50 mixture of unsymmetrical dimethyl hydrazine and hydrazine.

TABLE 6A-III
PHYSICAL AND THERMODYNAMIC PROPERTIES OF HYDROGEN AND OXYGEN

	Hydrogen*	Oxygen
Atomic Weight	1.008	16.000
Molecular Weight	2.016	32.000
Melting Point (^{o}R at 14.7 psia)	25.1	98.6
Boiling Point (^{o}R at 14.7 psia)	36.7	162.3
Critical Temperature (^{o}R)	59.38	278.92
Critical Pressure (Atm)	12.77	50.14
Critical Density (lb/ft^3)	1.90	26.8
Gas Density NTP (lb/ft^3)	0.0052	0.0827
Liquid Density (lb/ft^3)	4.37	71.3
Solid Density (lb/ft^3)	5.0 @25^{o}R	85.65 @ 98.1^{o}R
Viscosity of Liquid (Centipoise)	.0139 @ 36.4^{o}R	
Liquid Specific Heat, C_p (Btu/#-^{o}R	2.34 @ 36^{o}R	0.405 @ 162^{o}R
Gas Specific Heat, C_p (Btu/# -^{o}R) @ 540^{o}R & 14.7 psia	3.420	0.2199
Ratio of Specific Heats, (C_p/C_v) @ 540^{o}R & 14.7 psia	1.405	1.396
Gas Constant, R (ft lb/lb-^{o}R)	767.0	48.32
Thermal Conductivity (Btu/ft-hr-^{o}R)	681 @ 36^{o}R	0.00496 @ 162^{o}R
Heat of Vaporization (Btu/Lb.)	193 @ 36.7^{o}R	91.26 @ 162.3^{o}R
Heat of Fusion (Btu/Lb.)	25.2	5.98

*Values given are for equilibrium hydrogen, i.e., essentially parahydrogen at 36.7^{o}R and normal hydrogen at room temperature.

APPENDIX 6B
ENGINE SYSTEMS PARAMETRIC DATA

1.0 GENERAL

The purpose of this phase of the study was to establish engine systems parametric data consistent with the state of the art for the predicted time period of the LEM. These data were established based on information supplied by the engine manufacturers. Parametric data were developed only for the propellant combinations as selected in Appendix 6A. These were (a) nitrogen tetraoxide and Aerozine 50 (earth storable) and (b) liquid oxygen and liquid hydrogen (cryogenic).

Engine systems are defined as thrust chamber, pump, chamber accessories, and the components of the propellant flow system leading from the tankage. The propellant flow system is not ordinarily included in engine systems, but was so defined here to combine all components which are dependent on the thrust level. The specific data presented in this section are weight, performance, and thrust chamber dimensions.

2.0 SYSTEM WEIGHT DATA

The weight data have been organized by the functions listed below:

a. Thrust chambers
b. Pump and pump drive
c. Accessories
d. Propellant flow system
e. Engine systems

This organization facilitated the synthesis of various possible engine concepts. The data presented herein are only a portion of that reviewed during the study, however, it does cover the spectrum of systems which proved to be practical for this application. The final build-up of data into engine systems is presented as it was used to synthesize propulsion systems throughout the study.

2.1 Ablative Thrust Chambers

The ablative thrust chamber weight data obtained from the various engine manufacturers varied quite widely. Figure 6B-1 presents an envelope of data received from five companies for engines with identical characteristics. Because of this wide spread and the general lack of development experience, it was decided to use the higher weight values throughout this study.

Summary plots of the ablative thrust chamber weights selected for use in the study are presented in Figure 6B-2. The data were developed for chamber pressures of 100 and 150 psia. Chambers designed for pressures lower than 100 psia were not considered because of their large size and possible problems with structural rigidity. The concensus of manufacturers

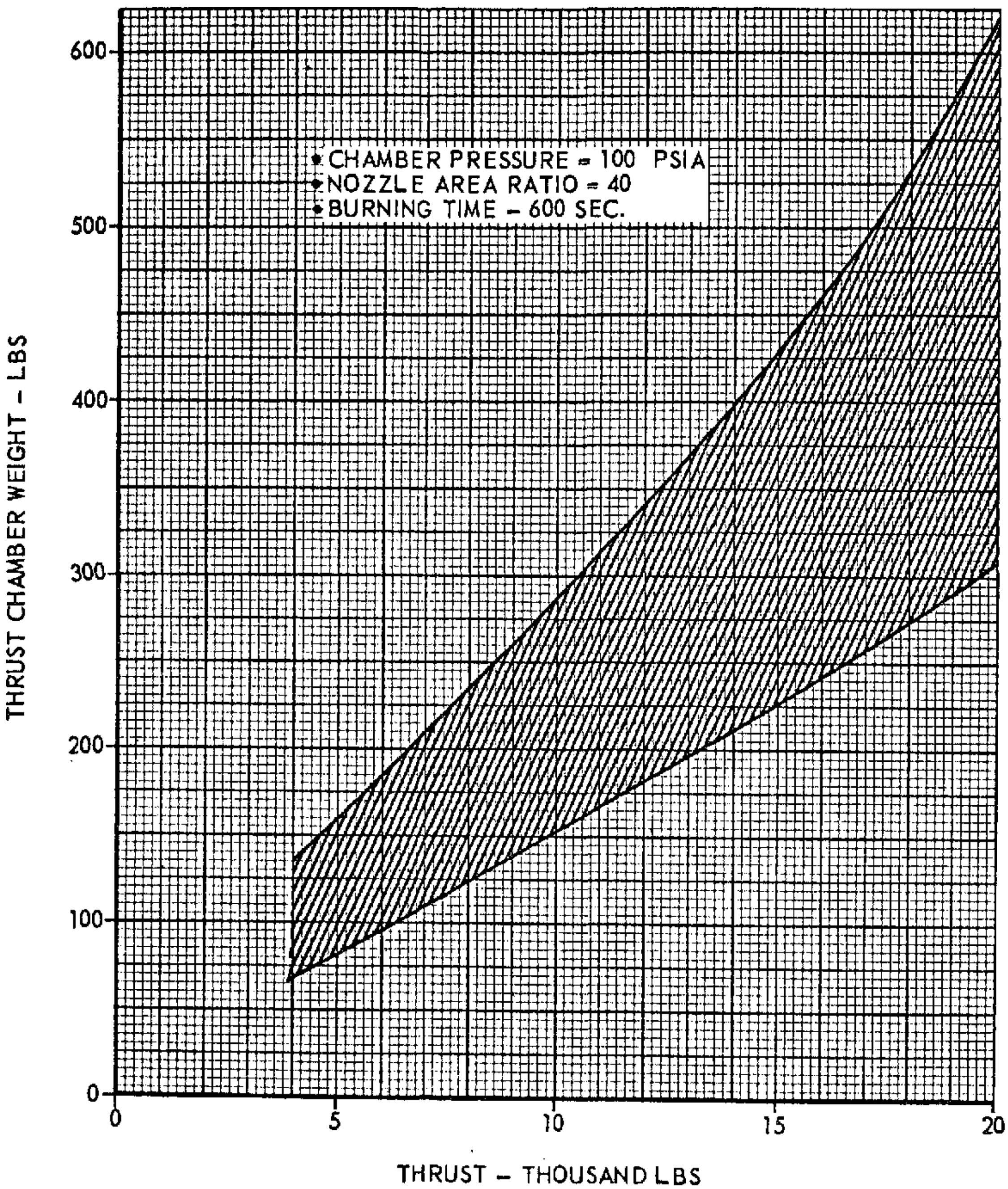


Figure 6B-1 COMPARATIVE ABLATIVE THRUST CHAMBER WEIGHTS

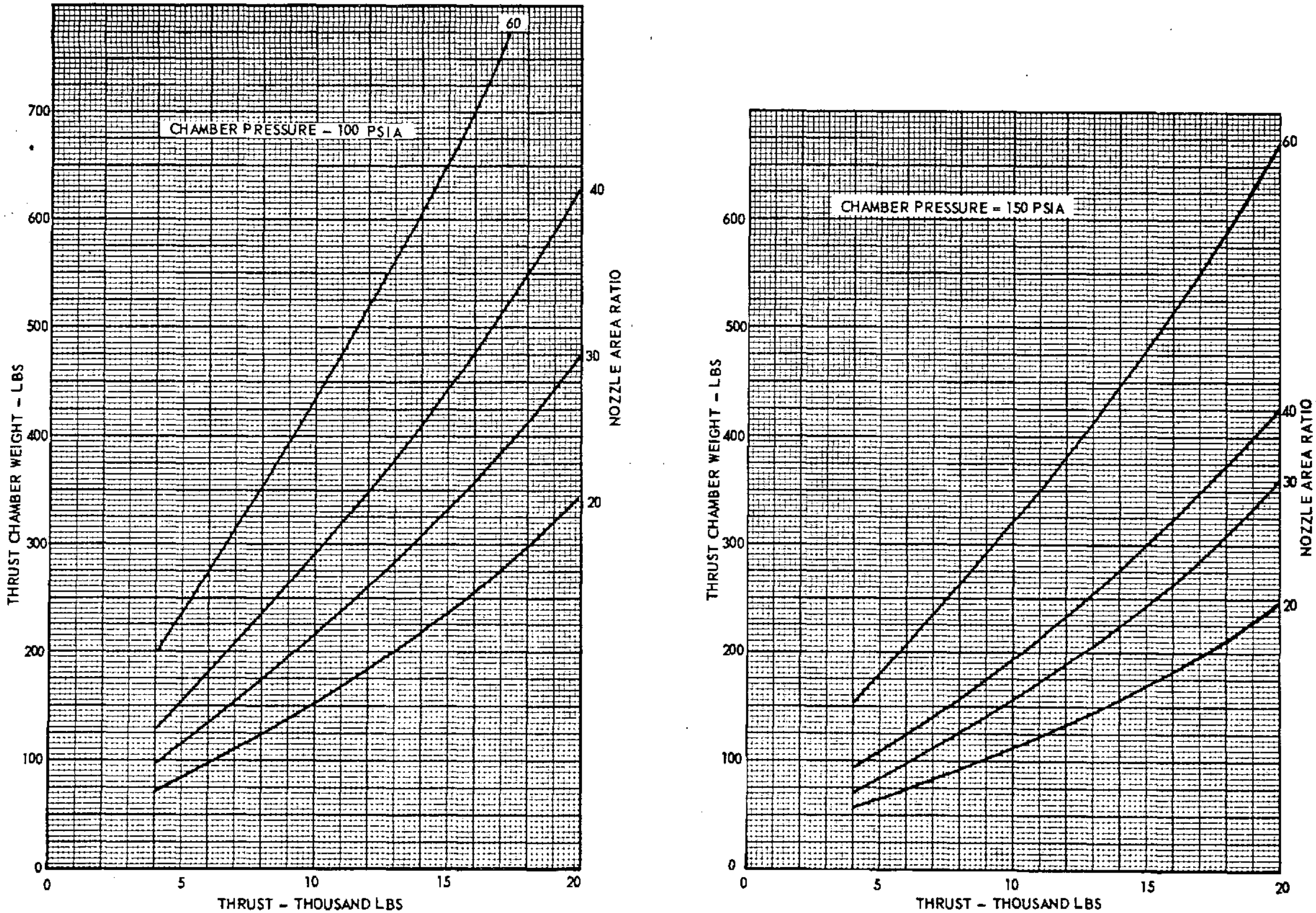


Figure 6B–2 ABLATIVE THRUST CHAMBER WEIGHT – BURN TIME 600 SECONDS regarding high chamber pressures indicated that 200 psia would be marginal relative to state of the art on ablatives. Therefore, it was decided to assure the chamber development feasibility by limiting the consideration to chamber pressures of 100 to 150 psia.

The chamber data are presented for a nominal design burning time of 600 seconds. Weights for other burning times can be obtained by adjusting the chamber weight using Figure 6B-3.

2.2 Regenerative Thrust Chambers

Regenerative thrust chamber weights were received from several sources and compared. In addition these weights were compared with data from engine systems already developed. The data presented in Figures 6B-4 and 6B-5 were very consistent from all sources as might be expected considering the large amount of experience in the industry. These data include both storable and cryogenic propellants.

The design chamber pressures of 100 and 300 psia were selected for this study. The lower pressure was selected as representative of a system designed for pressure feed, and the higher pressure as representative of a pump fed system.

2.3 Propellant Pumps and Drive

Cryogenic pump weight data presented are based on the RL-10 engine design. These data are presented in Figure 6B-6 and represent the expander cycle system.

Pump data for earth storable propellants is not presented, since the data received from the engine manufacturers were not sufficiently consistent to be considered reliable. Furthermore, pump fed storable propellant systems are not considered to be current state of the art and were therefore not evaluated in this study.

2.4 Engine Accessories

The engine accessory weights are presented in Figure 6B-7 for pump fed and pressure fed engine systems. It was necessary to distinguish between pump and pressure fed engine systems because the accessory weights are significantly different. They include, as applicable:

a. Propellant lines and valving
b. Pump drive exhaust ducting
c. Ignition
d. Controls and electrical connections

2.5 Propellant Flow System

Propellant flow system weights are presented in Figure 6B-8. The data presented include:

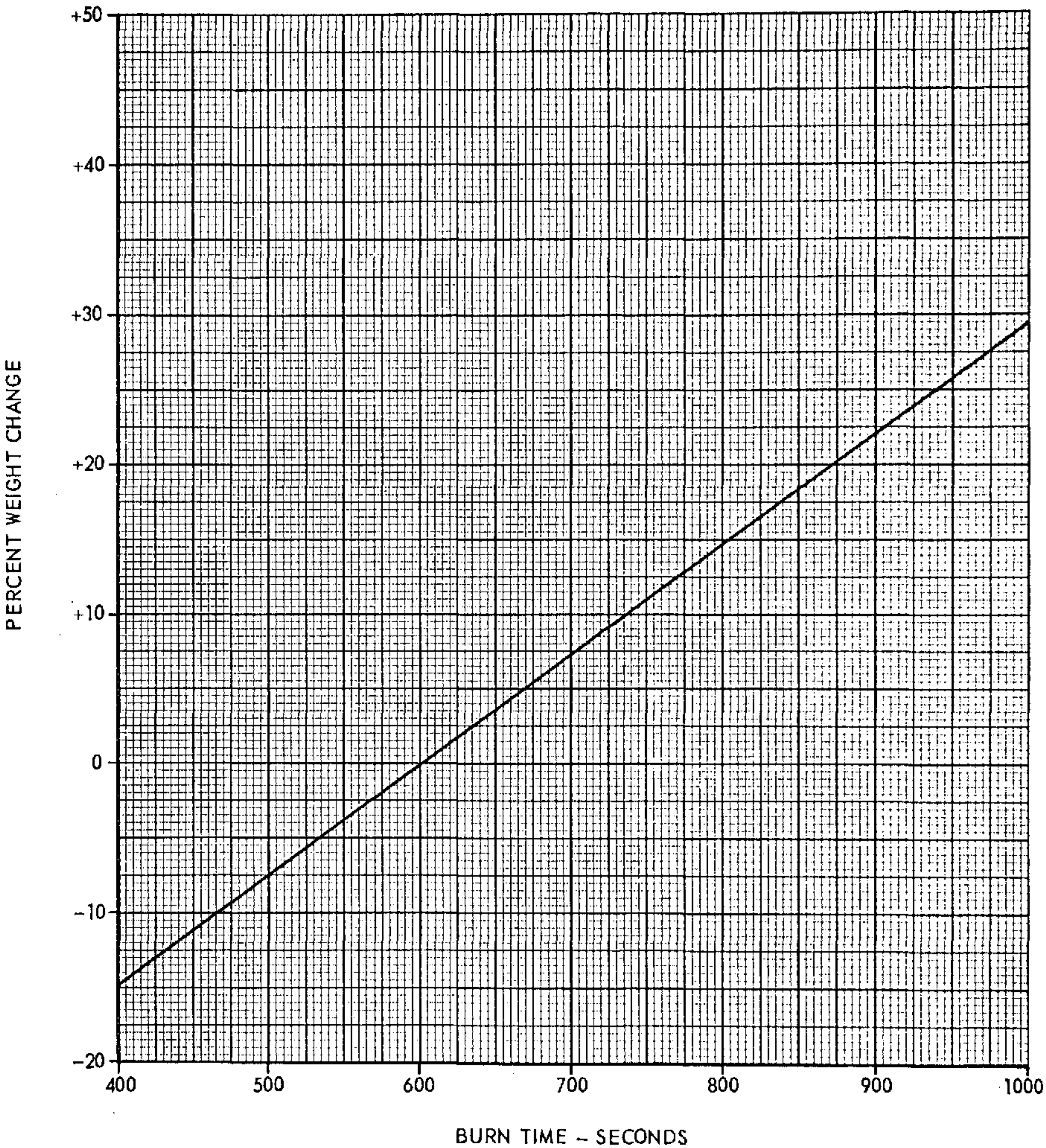


Figure 6B-3 DESIGN DURATION EFFECT ON ABLATIVE CHAMBER WEIGHT

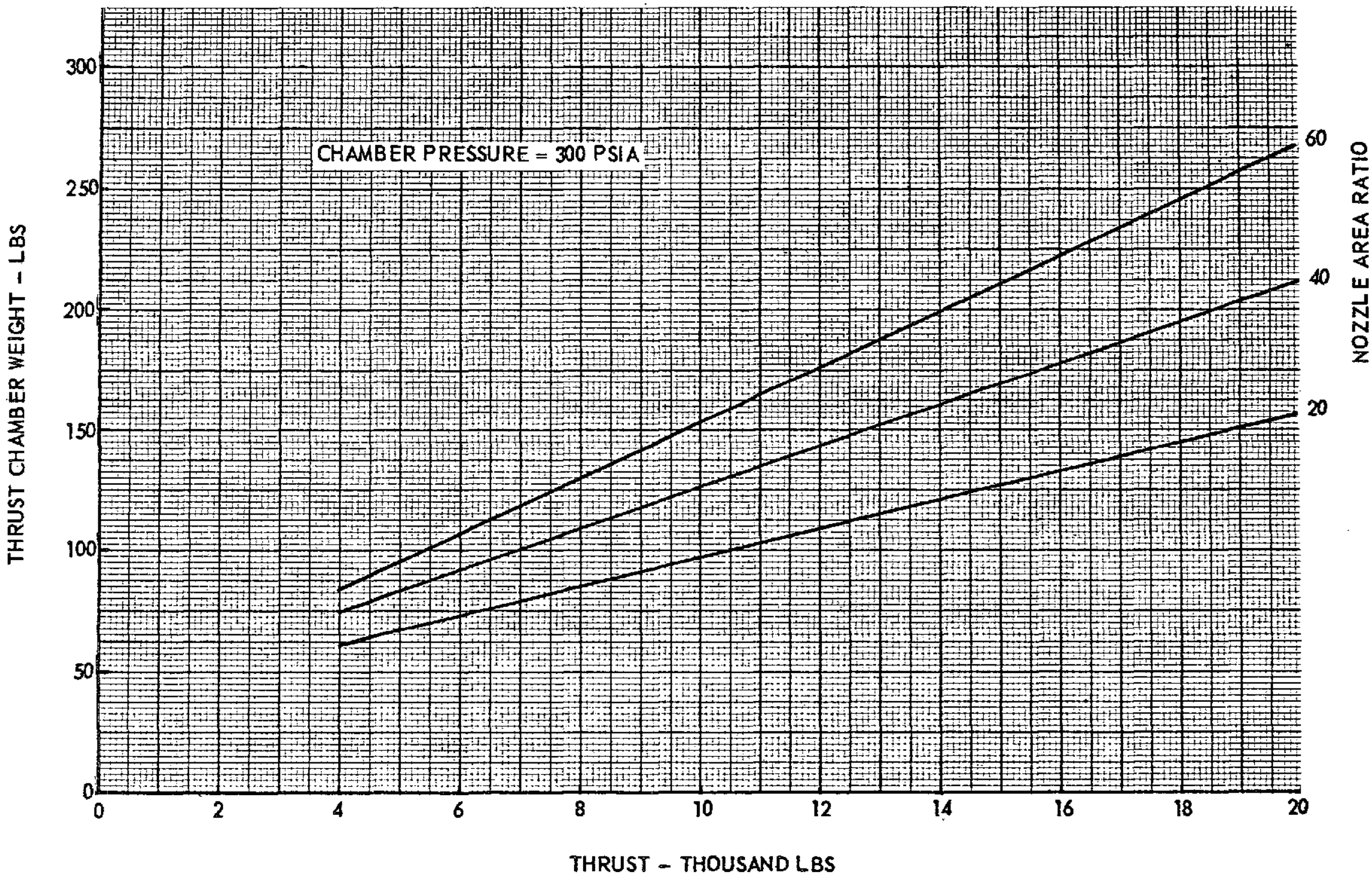


Figure 6B-4 REGENERATIVE THRUST CHAMBER WEIGHT - PUMP FEED

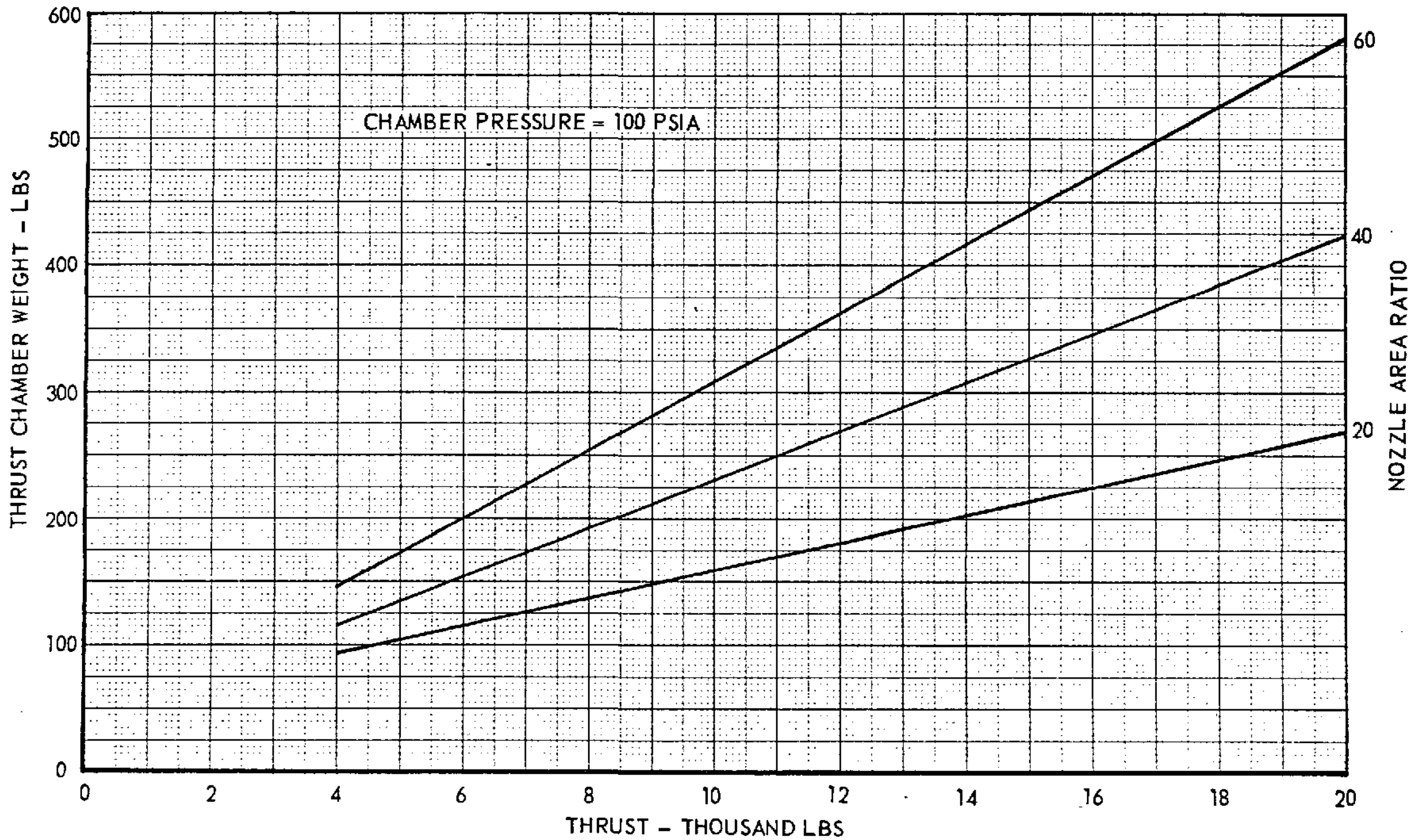


Figure 6B-5 REGENERATIVE THRUST CHAMBER WEIGHT – PRESSURE FEED

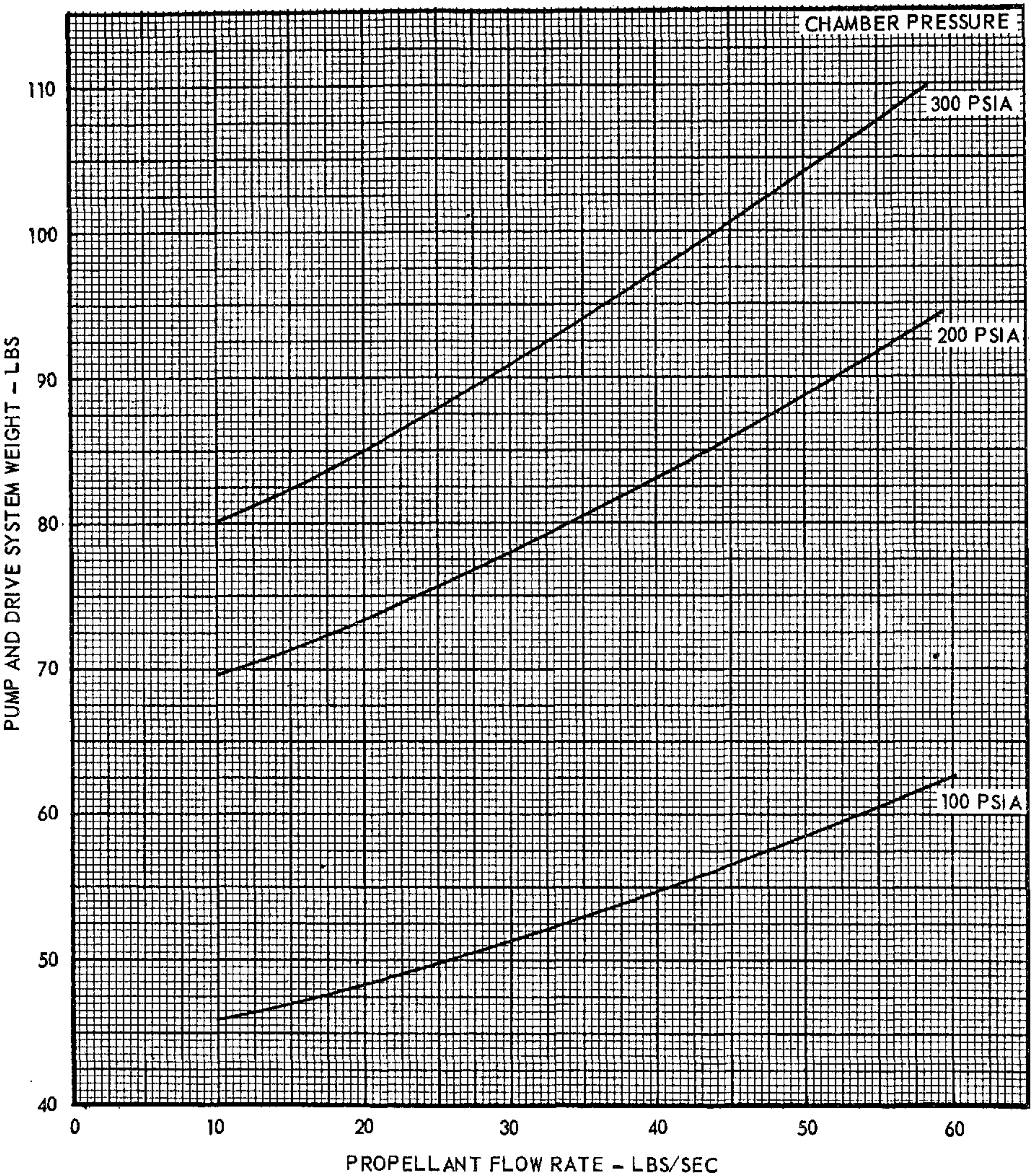


Figure 6B-6 CRYOGENIC PUMP AND DRIVE SYSTEM WEIGHT

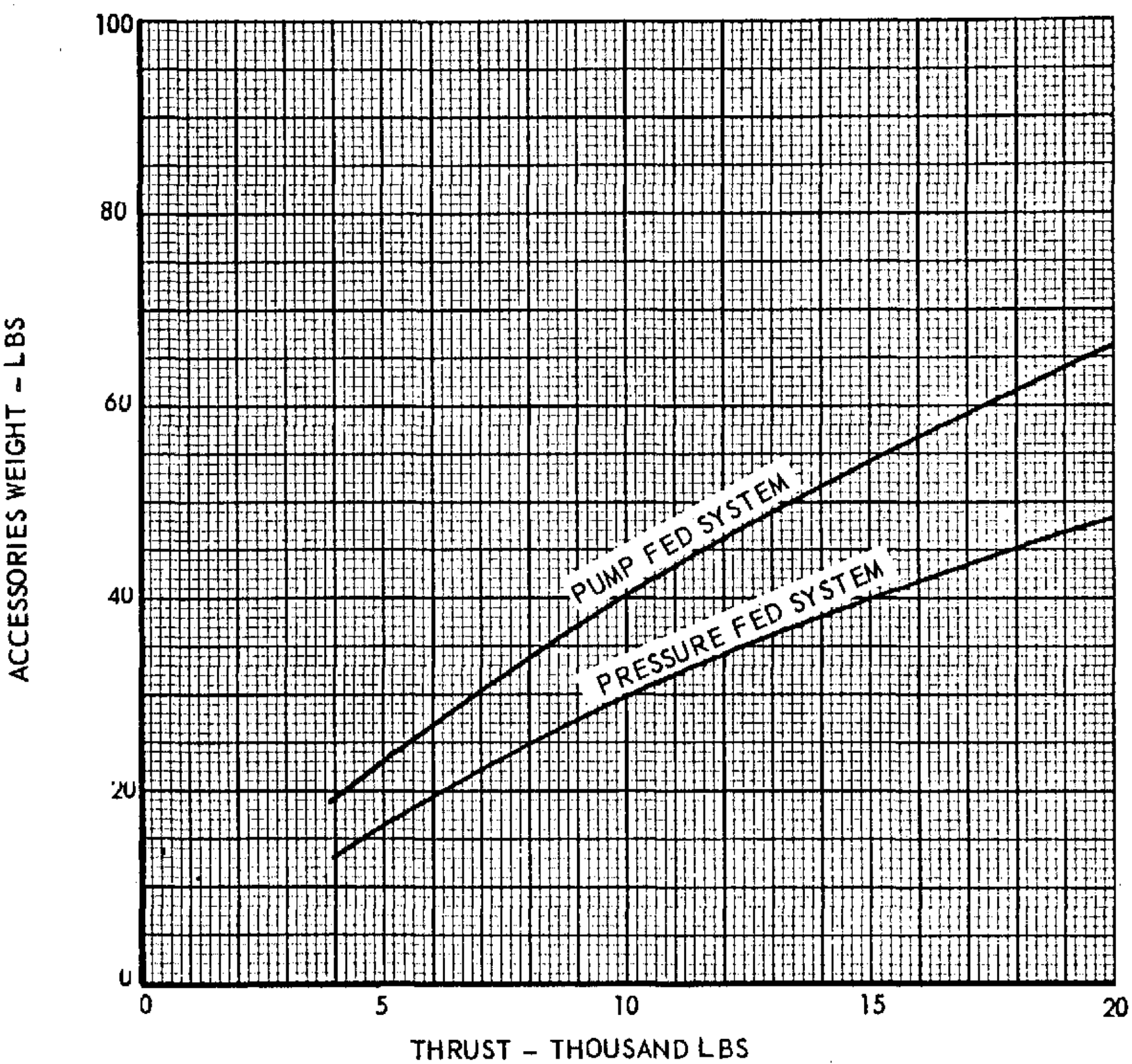


Figure 6B–7 ENGINE MOUNTED ACCESSORIES WEIGHT

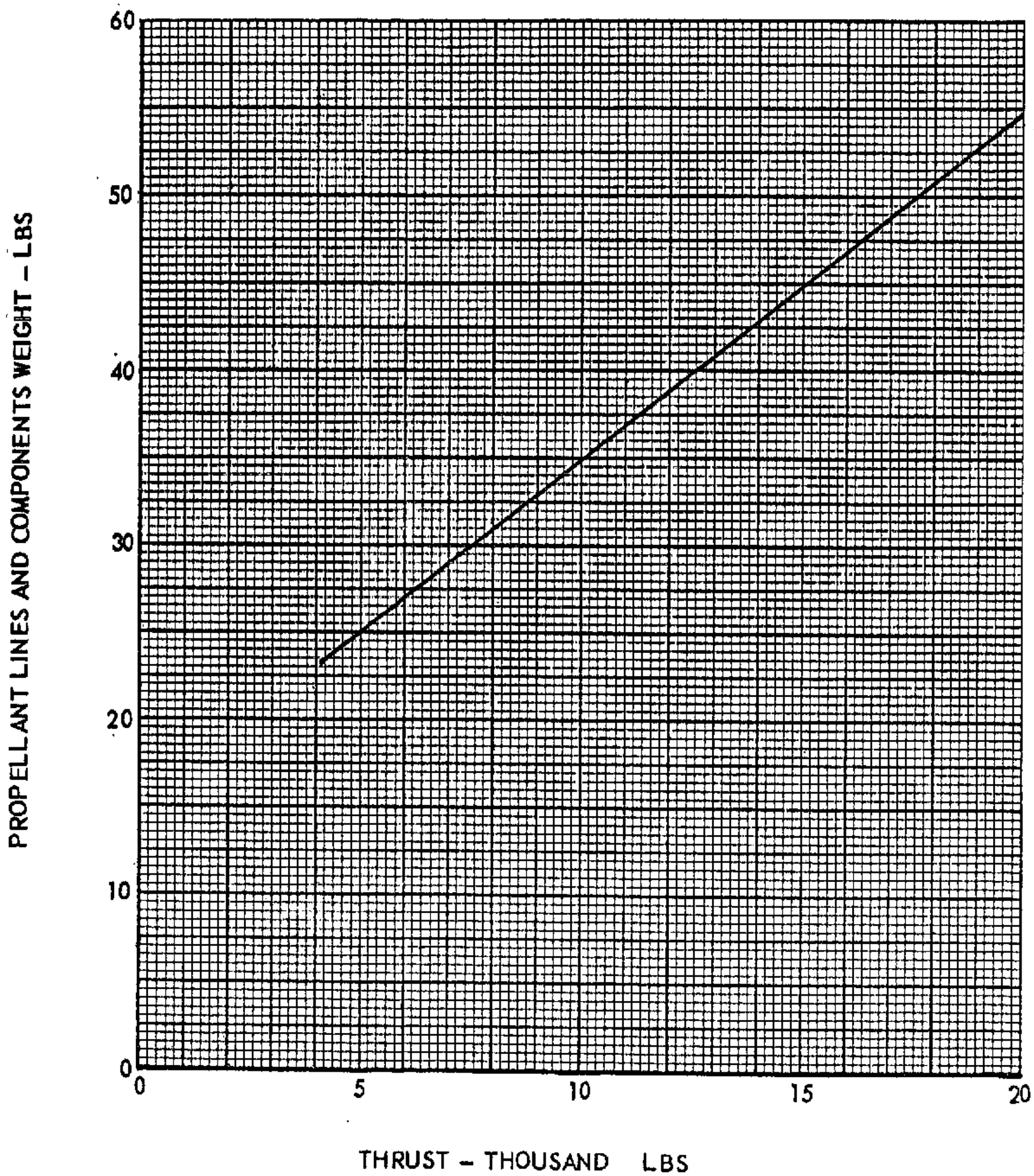


Figure 6B-8 PROPELLANT FLOW SYSTEM WEIGHT

a. Propellant feed lines and insulation
b. Valves
c. Filters
d. Sensing devices
e. Connections

The values presented are nominal because there can be considerable variation in line length, component weights, etc., depending on the vehicle arrangement and the system concept. The weights shown are considered conservative for earth storable feed systems and unconservative for cryogenic systems where line sizes and components are large. The values are used only for generating parametric systems. Final system weight determination requires detail weight evaluation of these components.

2.6 Systems

The weights for the engine systems, given the most detailed consideration in subsequent study work, were assembled from the above component data. These weights are presented for one and three thrust chamber pressure fed engine configurations using earth storable propellant in Figure 6B-9. Data is presented for cryogenic pump fed systems in Figure 6B-10 and cryogenic pressure fed systems in Figure 6B-11.

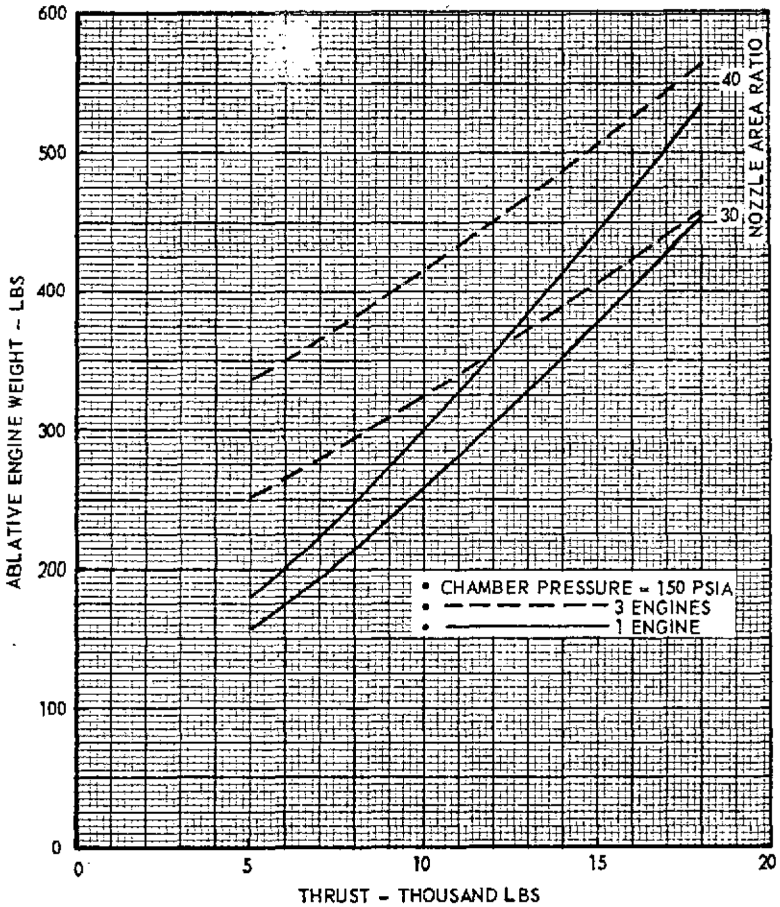
3.0 PERFORMANCE DATA

3.1 Earth Storable Engine Performance

The theoretical propellant specific impulse and the selected engine design specific impulse are shown on Figure 6B-12 for earth storable propellants. The ratio of design to theoretical is 0.945 at the mixture ratio for maximum performance. Test data have been obtained in some cases which show ratios as high as 0.98. Thus, it is felt that the performance data used in this study are conservative. A design mixture ratio of 2.0 was selected for maximum performance and yields a specific impulse of 320 with a nozzle area ratio of 40.

The throttled performance of fixed and variable injector chambers is shown on Figure 6B-13 as a percent of design specific impulse.

The fixed injector performance levels represent the judgement and experience of the engine manufacturers in developing low pressure drop injectors which maintain stable combustion. It was determined that injector pressure drops as low as 6 psi would give the performance shown at the maximum throttle ratio of seven. This could only be true, however, if the total of chamber pressure and injector pressure drop was held to a value somewhat above the vapor pressure of the propellants at their injection temperatures. High throttle ratios then dictate high injection pressures at full thrust, and a throttle ratio of seven was considered a practical maximum for consideration in this study. Care was taken in the study not to exceed these limitations.



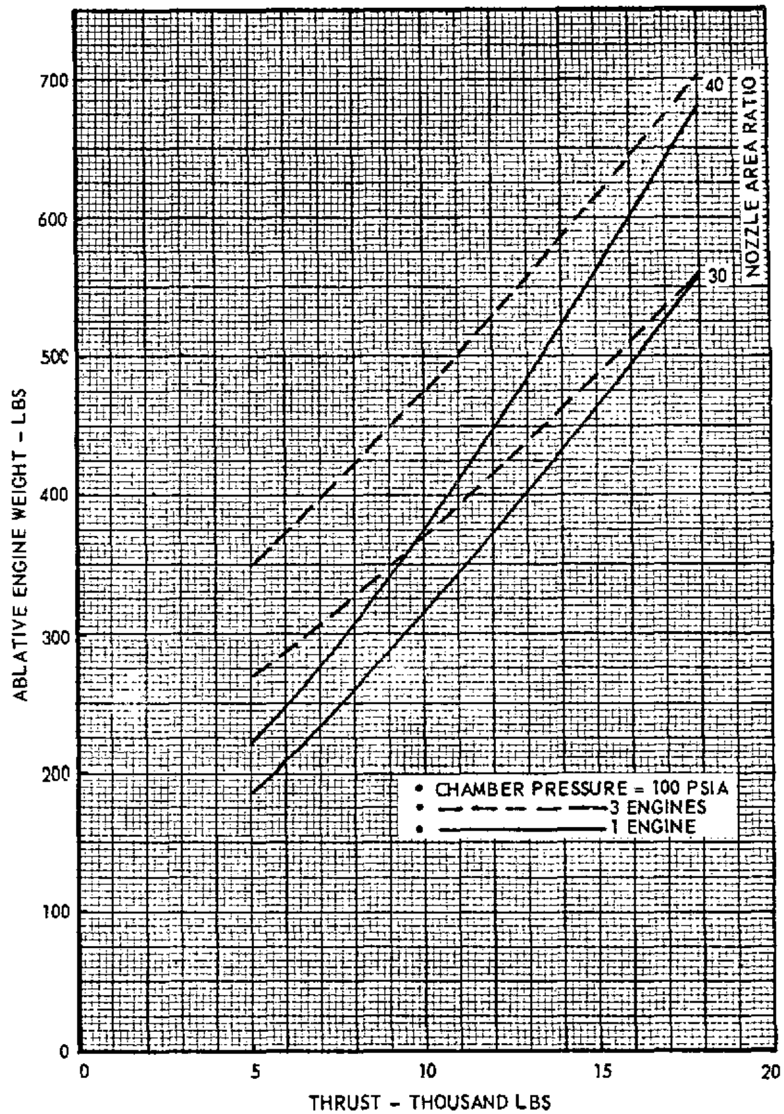


Figure 6B-9 ABLATIVE ENGINE WEIGHT

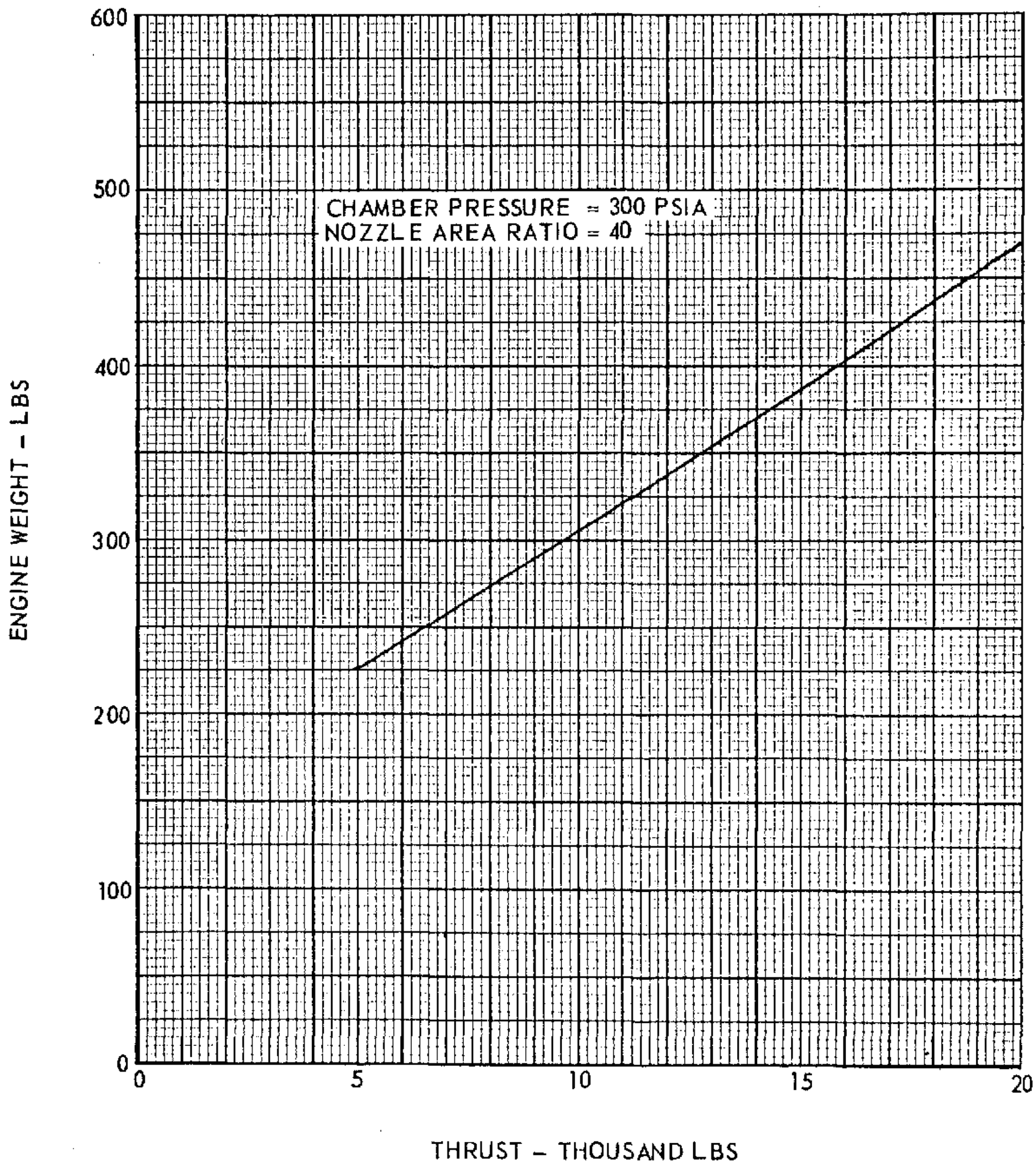


Figure 6B-10 CRYOGENIC PUMP FED ENGINE WEIGHT – REGENERATIVE

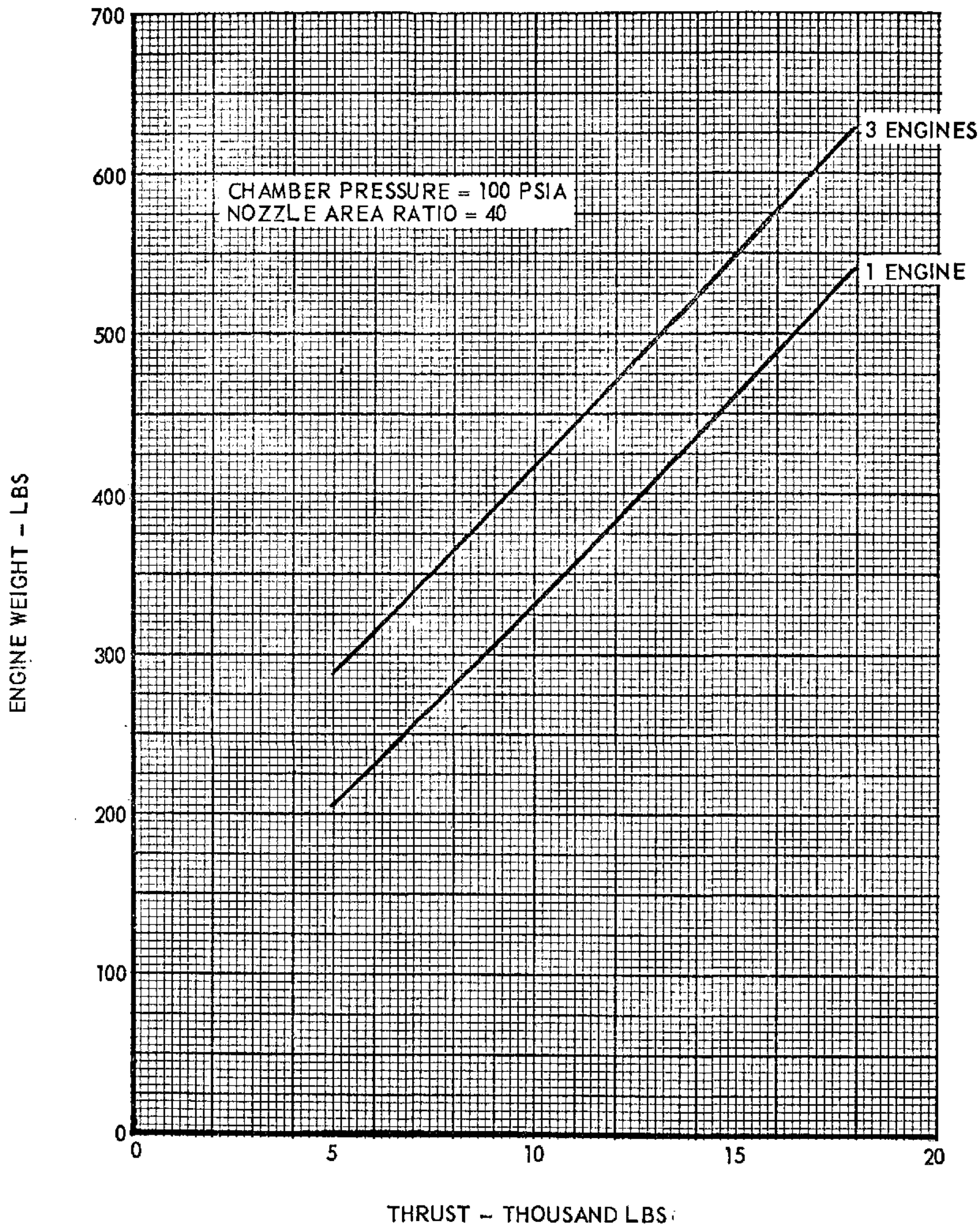


Figure 6B-11 CRYOGENIC PRESSURE FED ENGINE WEIGHT - REGENERATIVE

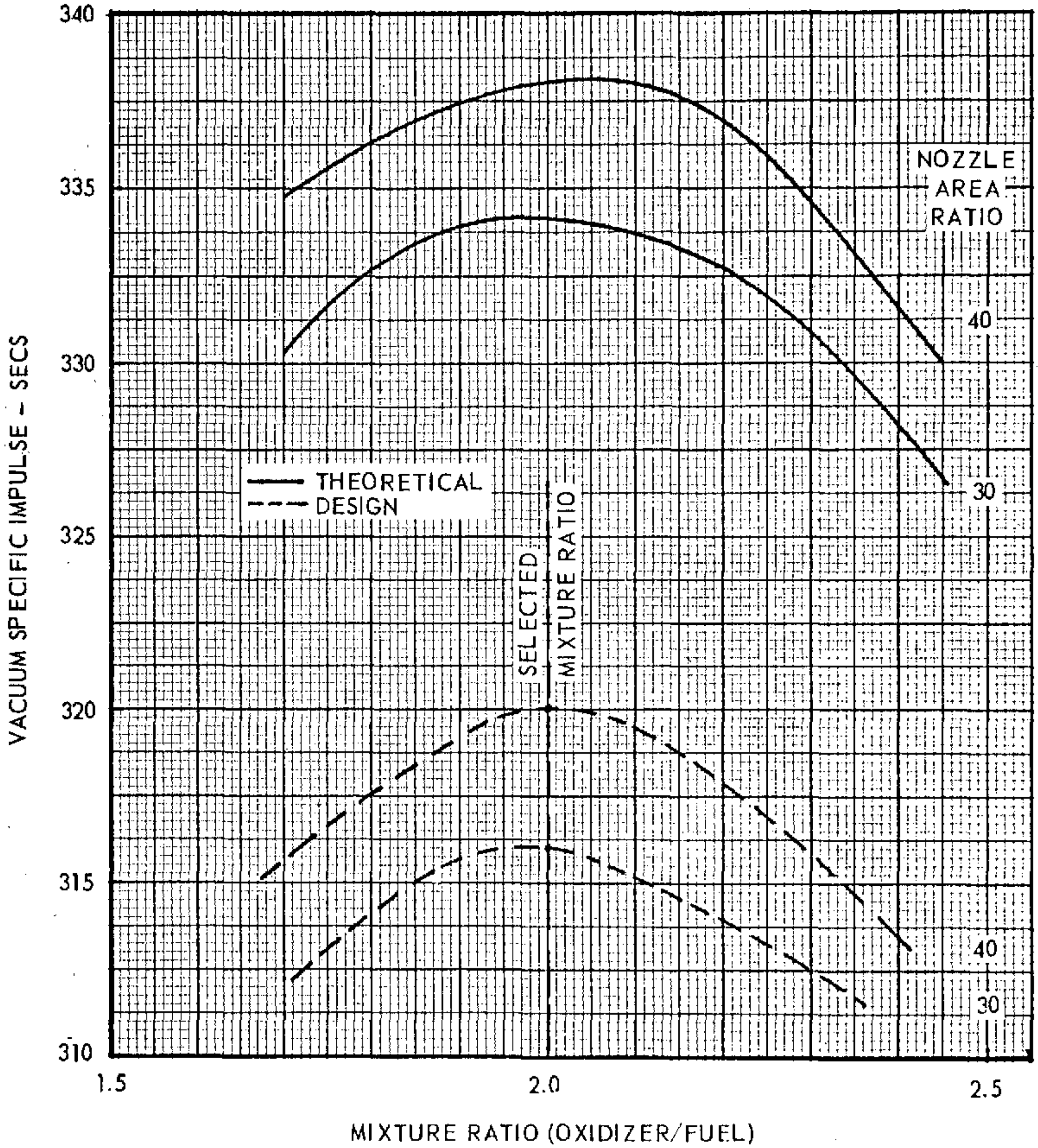


Figure 6B-12 VACUUM SPECIFIC IMPULSE - EARTH STORABLE

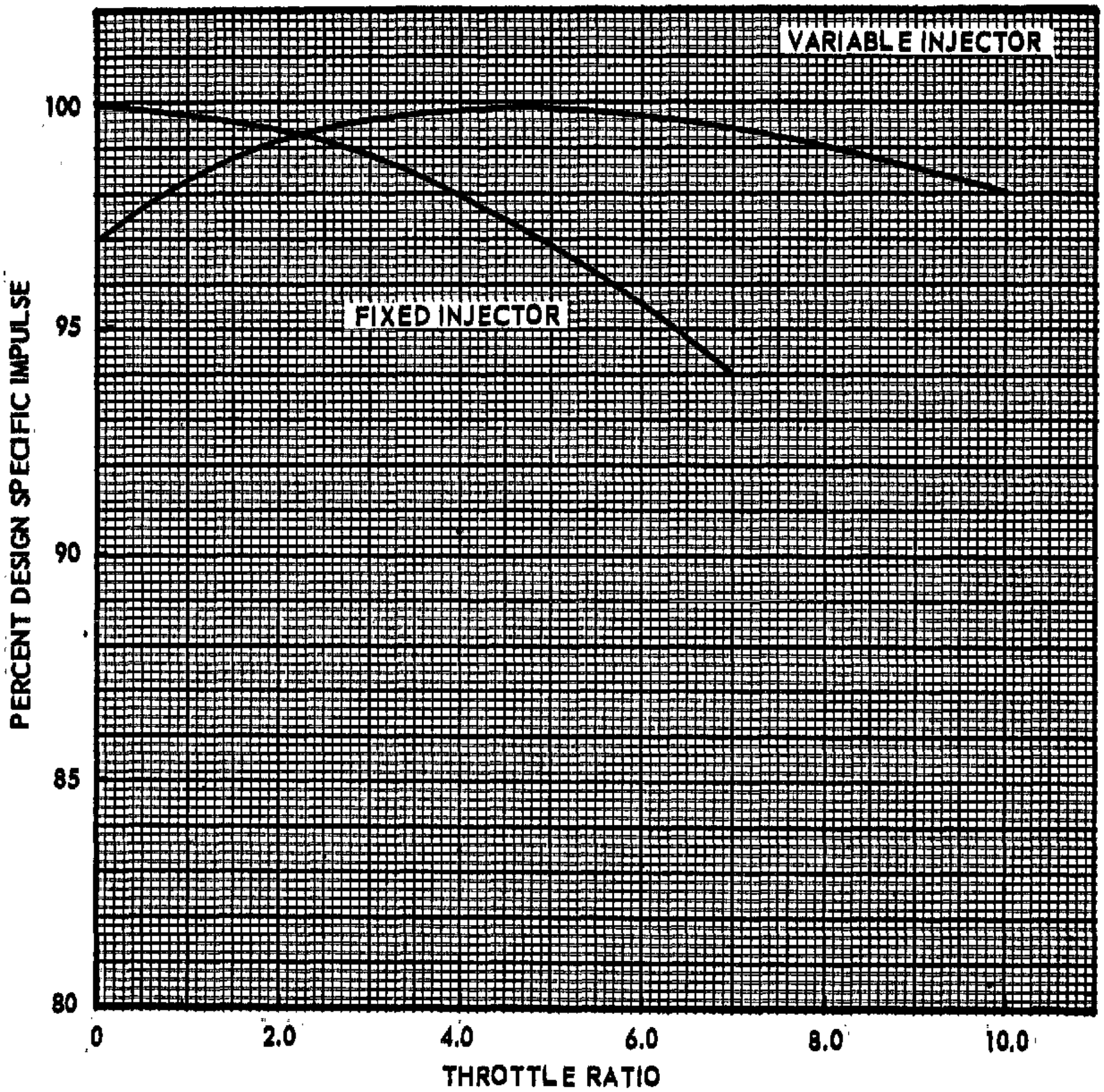


Figure 6B-13 THRUST CHAMBER PERFORMANCE - THROTTLED EARTH STORABLE

Variable injectors are still in the experimental stage. It was difficult to ascertain what performance characteristics might accompany a successful design. However, data from several experimental configurations were examined and a probable performance trend was estimated.

3.2 Cryogenic Engine Performance

The theoretical and engine design specific impulses are shown in Figure 6B-14 for various mixture ratios. The design curve marked "initial" shows the performance used in the preliminary studies. This was determined to be too conservative from RL-10 engine data and the "final" study evaluation was done using the upper curve. A mixture ratio of 5 was selected as a good compromise and gives a design specific impulse of 430. Higher mixture ratios would have been desirable to reduce system weight and size. However, fuel cooling is used in regenerative chambers and higher mixture ratios create a difficult cooling problem.

The throttled performance for a regenerative thrust chamber is shown in Figure 6B-15. The data came directly from experience with the RL-10 engine. Throttling performance for a non-regenerative chamber could not be established. Injection problems associated with both oxygen and hydrogen at their fluid temperatures have not been explored sufficiently to establish the throttling capability.

4.0 THRUST CHAMBER DIMENSIONS

4.1 Ablative Thrust Chambers

The thrust chamber dimensions used in LEM vehicle design studies are presented as Figures 6B-16, 6B-17 and 6B-18. These data, obtained from engine manufacturers, showed good agreement and are consistent with the thrust chamber weights presented in this Section.

4.2 Regenerative Thrust Chambers

(a) Pump Feed

Pratt and Whitney thrust chamber dimensions for the RL-10 were used in all cryogenic pump fed regenerative system design studies. This was done because a review of the development potential indicated that only modifications to the RL-10 could be developed within the LEM time period. This being the case the engine dimensions would not significantly change within the thrust range under consideration.

(b) Pressure Feed

The ablative thrust chamber dimensions presented above were used in vehicle design studies involving pressure fed regenerative thrust chambers. These dimensions are applicable since the only difference in the two types of chambers is that an ablative combustion chamber is slightly larger than a regenerative chamber. Hence, the dimensions used for regenerative pressure fed thrust chambers were conservative.

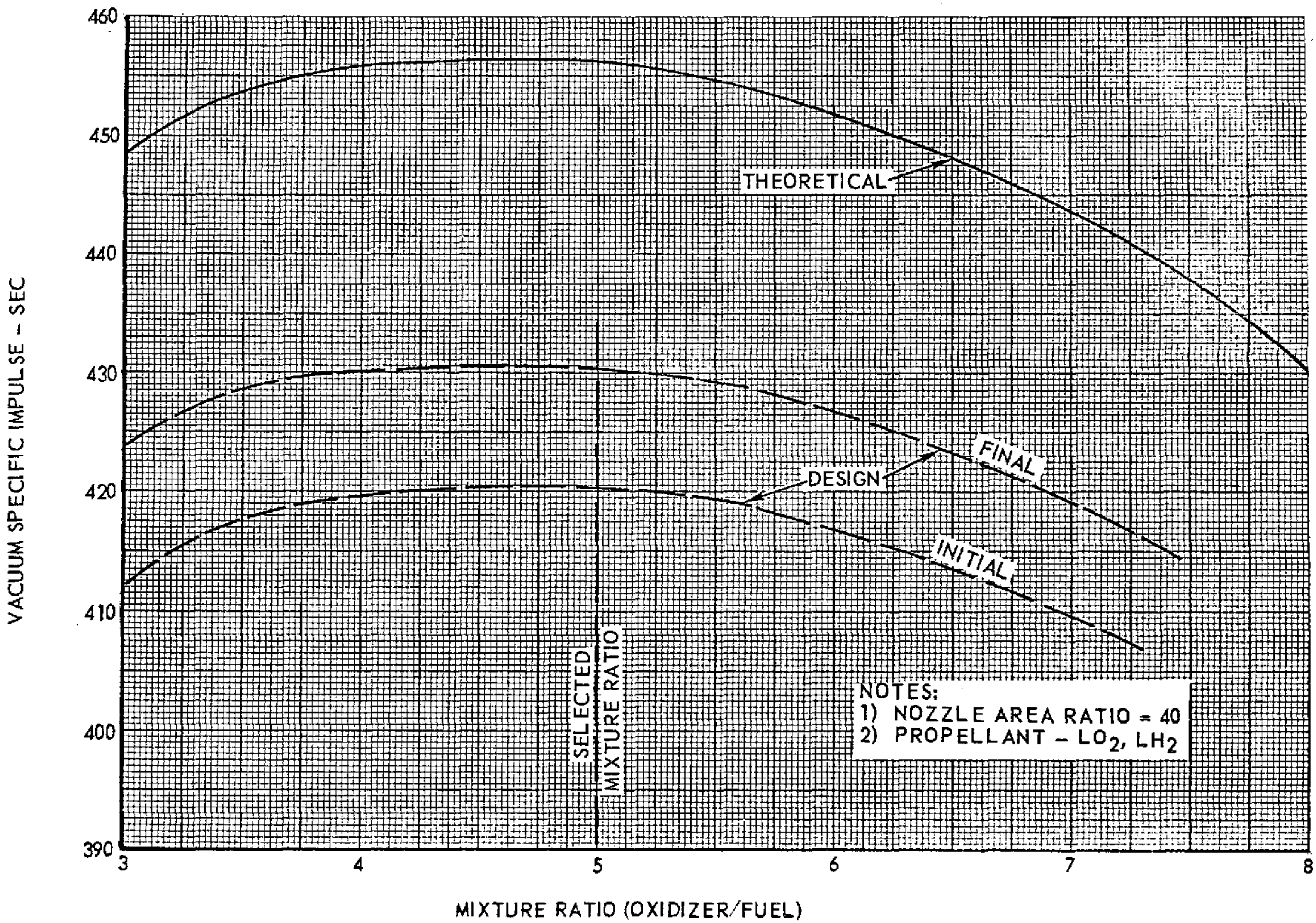


Figure 6B-14 VACUUM SPECIFIC IMPULSE - CRYOGENIC

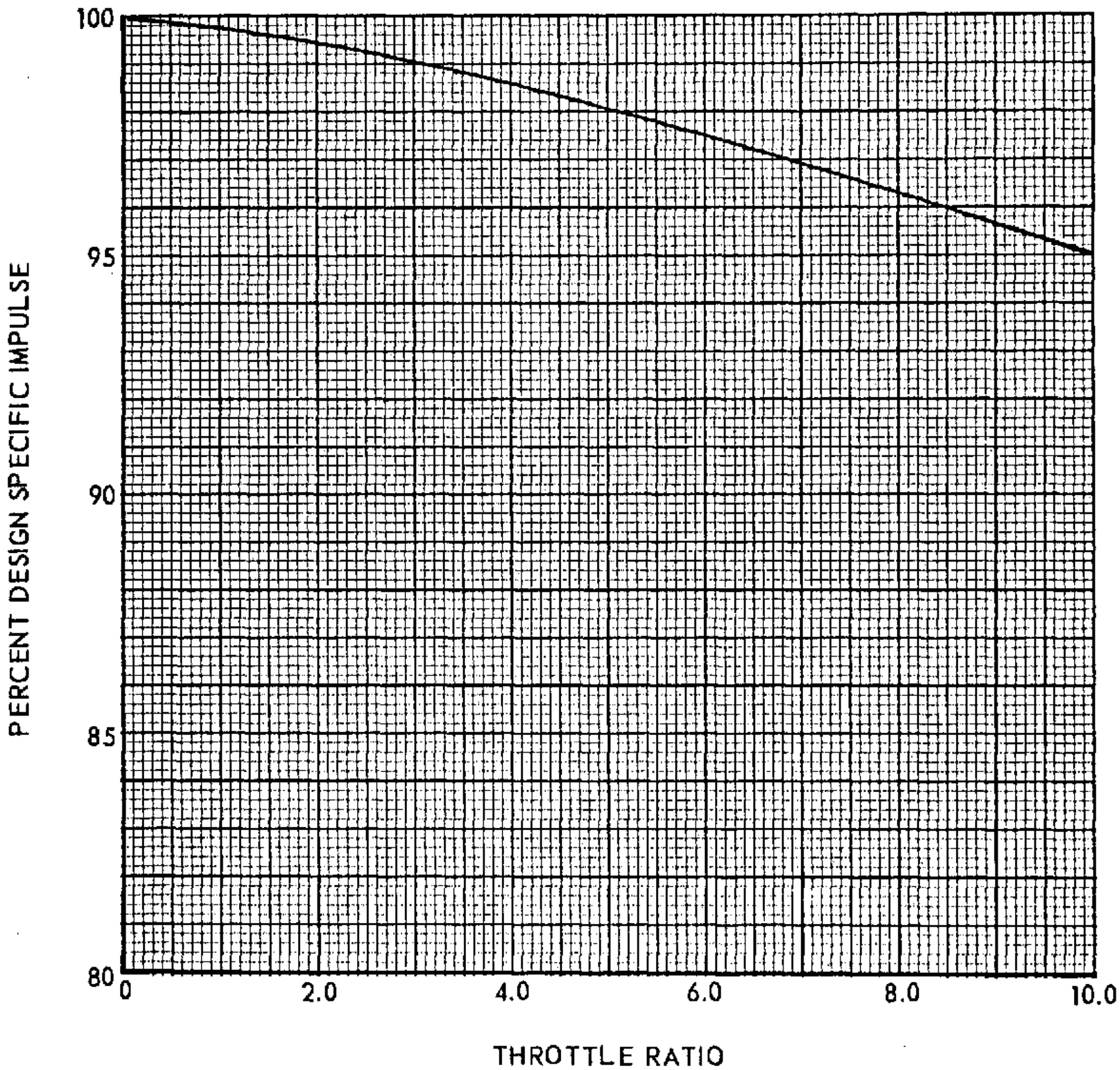


Figure 6B-15 REGENERATIVE THRUST CHAMBER PERFORMANCE – THROTTLED CRYOGENIC

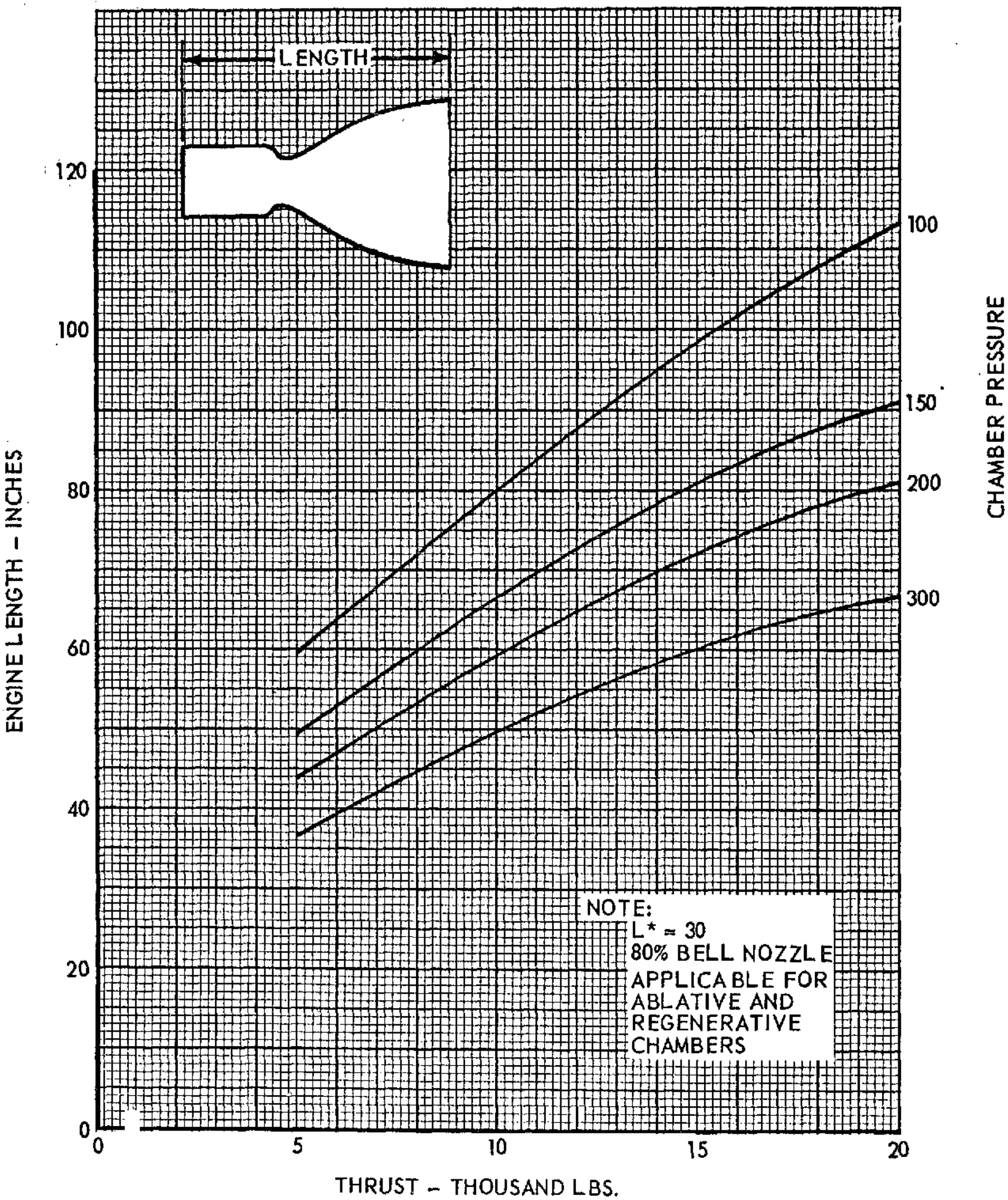


Figure 6B-16 ENGINE LENGTH - NOZZLE AREA RATIO 40

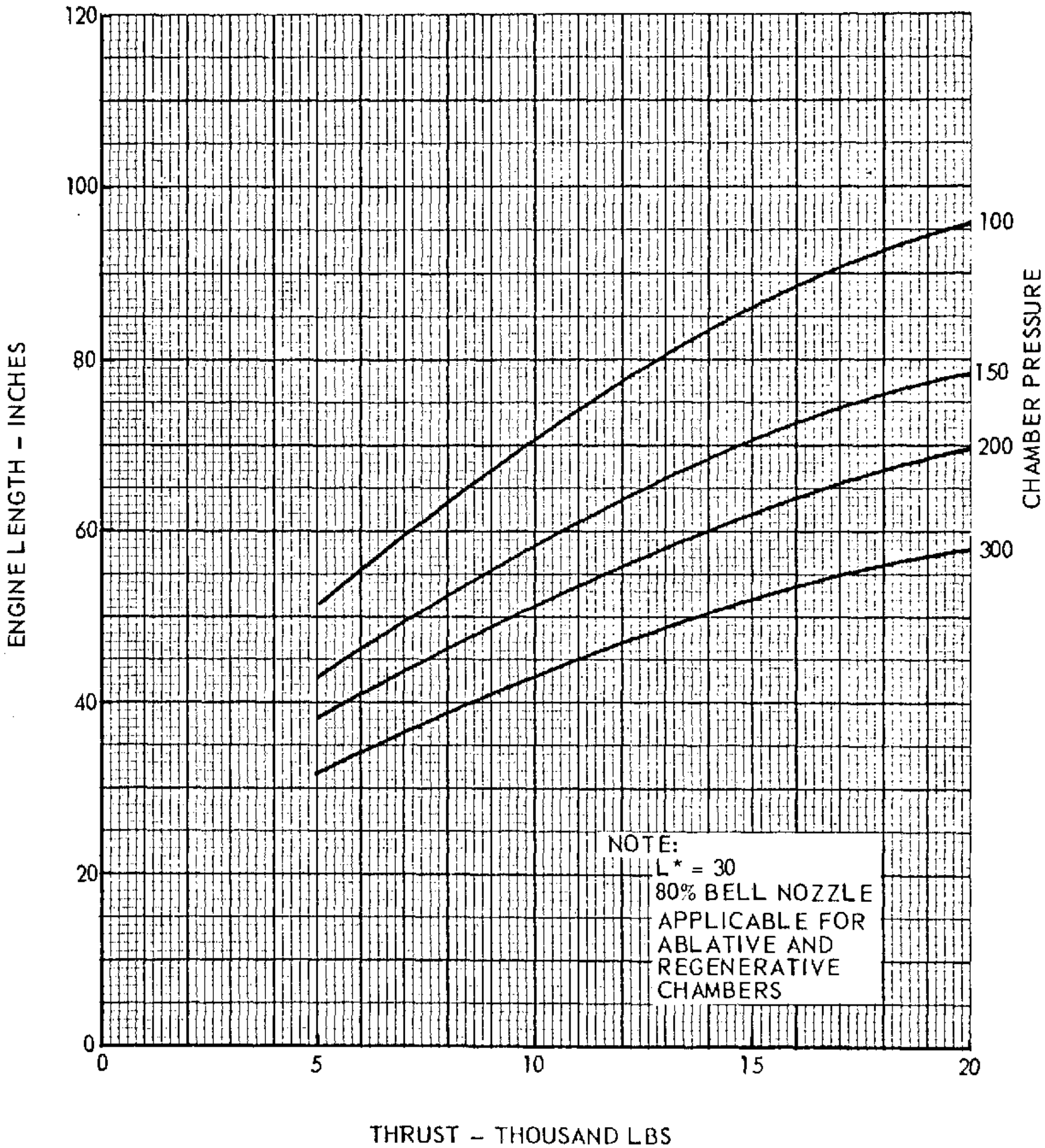


Figure 6B–17 ENGINE LENGTH – NOZZLE AREA RATIO 30

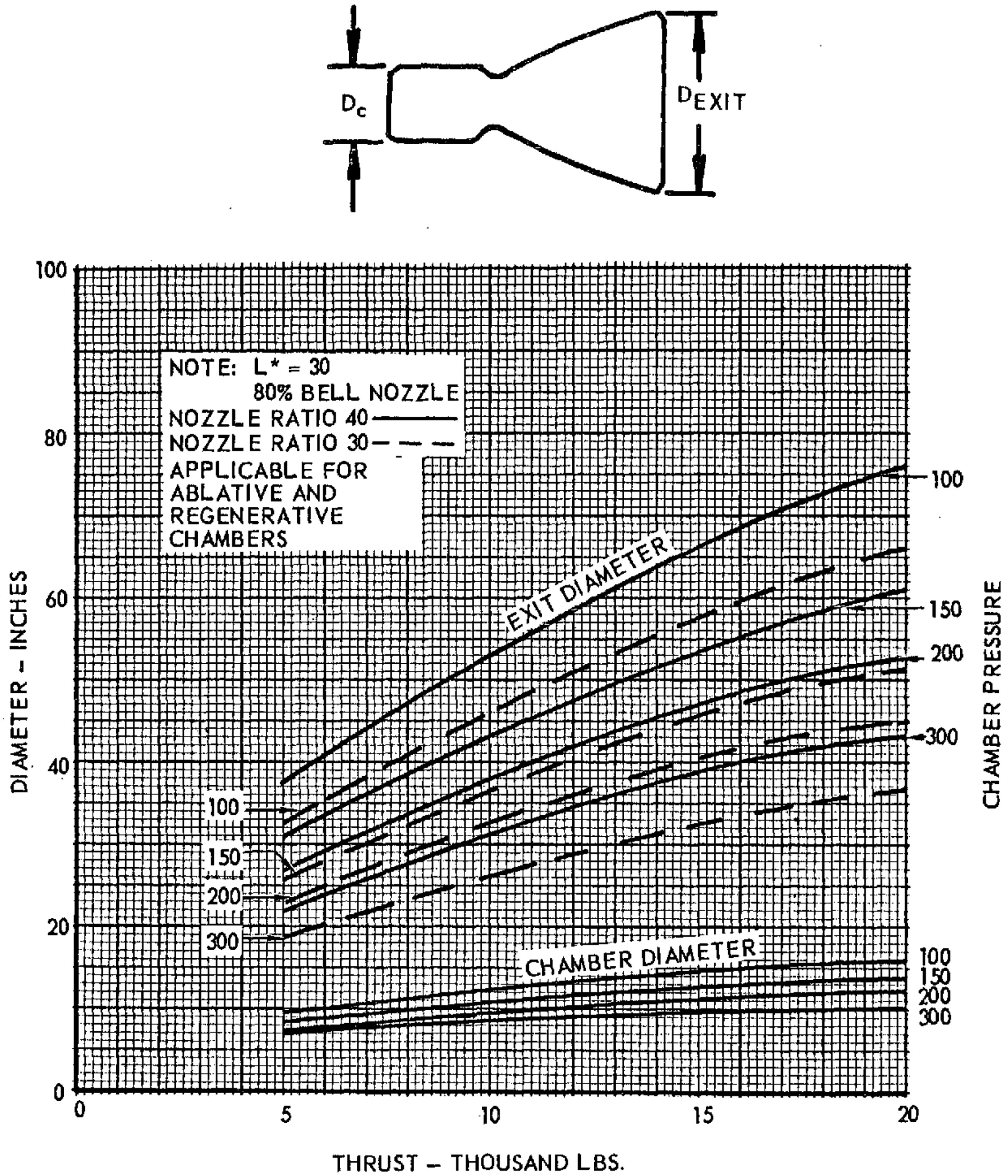


Figure 6B-18 NOZZLE EXIT AND CHAMBER DIAMETERS

APPENDIX 6C

PROPELLANT STORAGE AND PRESSURIZATION

1.0 GENERAL

This appendix presents a summary of the LEM studies on propellant storage and pressurization. The objective of these studies was to examine potential storage and pressurization systems and to select the most practical system for the LEM. Since the propellant combination was a variable in the study, both cryogenic and storable propellants were considered. The cryogenic propellant combination studied was liquid oxygen and liquid hydrogen and the storable propellant combination studied was nitrogen tetroxide (N_2O_4) and Aerozine-50. The selection criteria used for evaluating potential storage and pressurization systems for these propellants are listed below in the order of importance assigned:

(a) Reliability
(b) Development status
(c) Weight
(d) Configuration (shape and size)

The evaluation of propellant storage and pressurization systems for the LEM was accomplished by examining the three major design considerations separately. These three major considerations are:

(a) Propellant tanks and tank supports
(b) Propellant thermal protection
(c) Propellant tank pressurization

These three areas are considered separately in the following paragraphs.

2.0 PROPELLANT TANKS AND TANK SUPPORTS

The basic requirement of the propellant tankage is to contain and support that quantity of propellant necessary to accomplish the mission. The propellant quantity required comprises approximately 80 to 90 per cent of the total propulsion system weight. Therefore, the tankage configuration is an important design consideration from the standpoint of overall propulsion system weight.

The major problems associated with determining a minimum weight combination of propellant tanks and supports are: (1) choice of materials which are compatible with the selected propellants and (2) determination of the best shape, size, and number of tanks. A detailed analysis of these two problems is presented in the Vehicle Structure Section (Section 2, Volume II). A summary of the method of approach and important conclusions is presented below.

The initial step in establishing a recommended tank design is to select materials that are compatible with the chosen propellants and are suitable from the manufacturing standpoint. The possible materials must then be evaluated to determine the material which results in the minimum weight tankage system. Based on this type evaluation the following materials were selected for the LEM:

Propellant	Tank Material
Liquid oxygen	2219 aluminum
Liquid hydrogen	5 Al - 2.5 Sn titanium
Aerozine-50	6 Al - 4V titanium
N_2O_4	6 Al - 4V titanium

After selecting the tank materials a study was required to establish the most desirable shape, size, and number of tanks. Major considerations included in this study are:

(a) Minimum gage required for fabrication
(b) Micrometeoroid protection
(c) Overall vehicle design requirements
(d) Maximum propellant utilization
(e) Hydrostatic pressures induced by earth boost
(f) Tank operating pressures
(g) Baffling required to control propellant slosh and swirl

As a result of the analysis of the above problems (Section 2.0, Volume II), a number of pertinent conclusions were drawn. These are tabulated below.

(a) At least two fuel and two oxidizer tanks will be required. This arrangement permits propulsion system staging, provides a low vehicle length to diameter ratio and reduces the C.G. control problem.

(b) Spherical tanks provide the minimum weight tankage configuration for storable propellants. Cylindrical tanks are lighter for cryogenic propellants because spherical hydrogen tanks will not fit within the C-5 fairing envelope without producing a significant increase in fairing weight. If cylindrical hydrogen tanks are used, the resulting tank support structural requirements make cylindrical oxygen tanks also desirable.

(c) The minimum material thickness required for fabrication is equal to or greater than the thickness required for micrometeroid protection.

(d) The hydrostatic pressure induced by earth boost may be neglected. This is permissible since the hydrostatic pressure is significantly lower than the pressure required for propellant feed.

The cryogenic and storable propellant storage system weights (tanks, supports and insulation) are presented in Figures 6C-1 and 6C-2 as a function of propellant weight.

3.0 PROPELLANT THERMAL PROTECTION

The LEM propellants must be loaded prior to earth launch and must remain stored until used during lunar descent and lunar launch. The

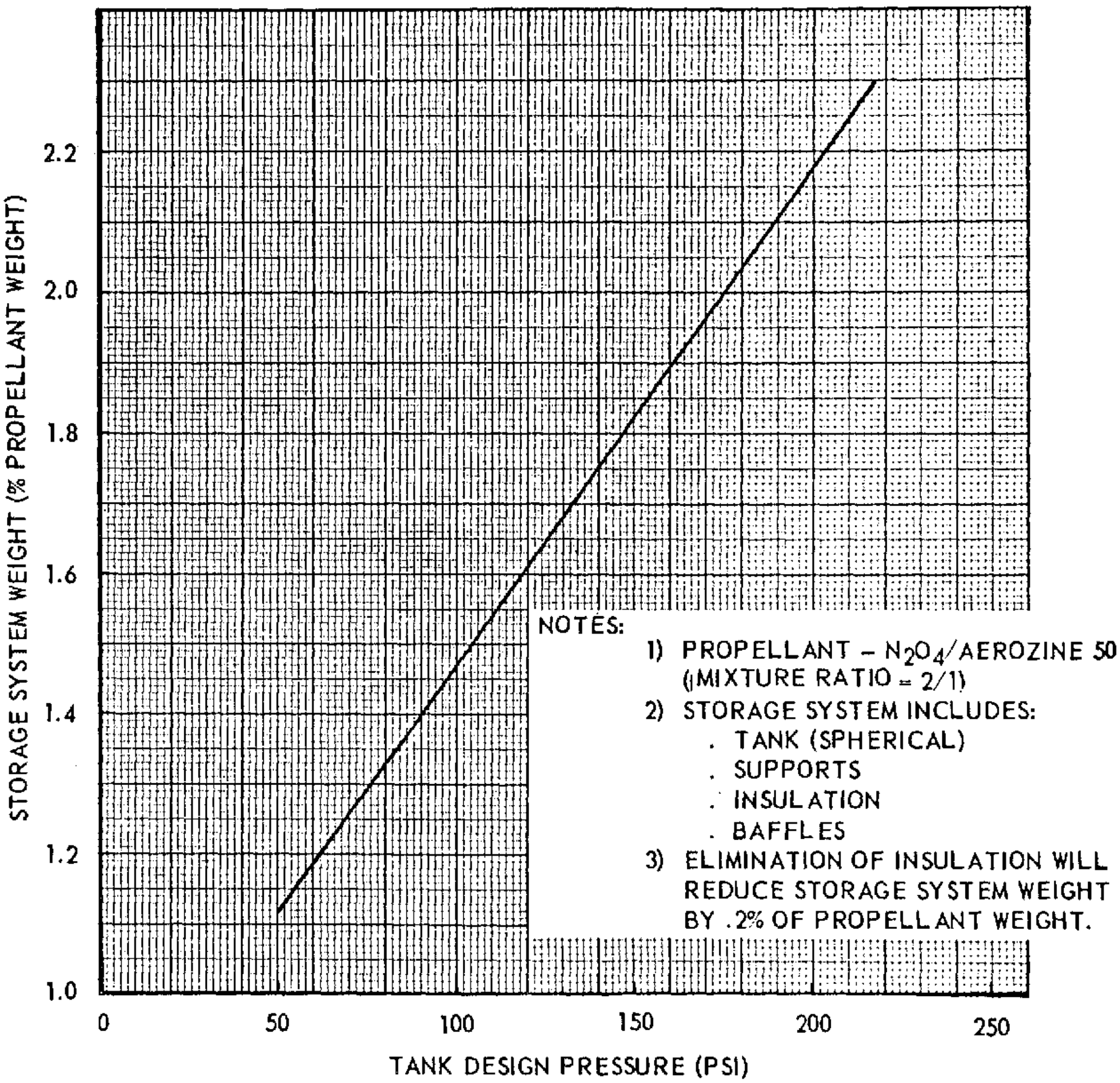


Figure 6C–1 STORABLE PROPELLANT STORAGE SYSTEM WEIGHTS

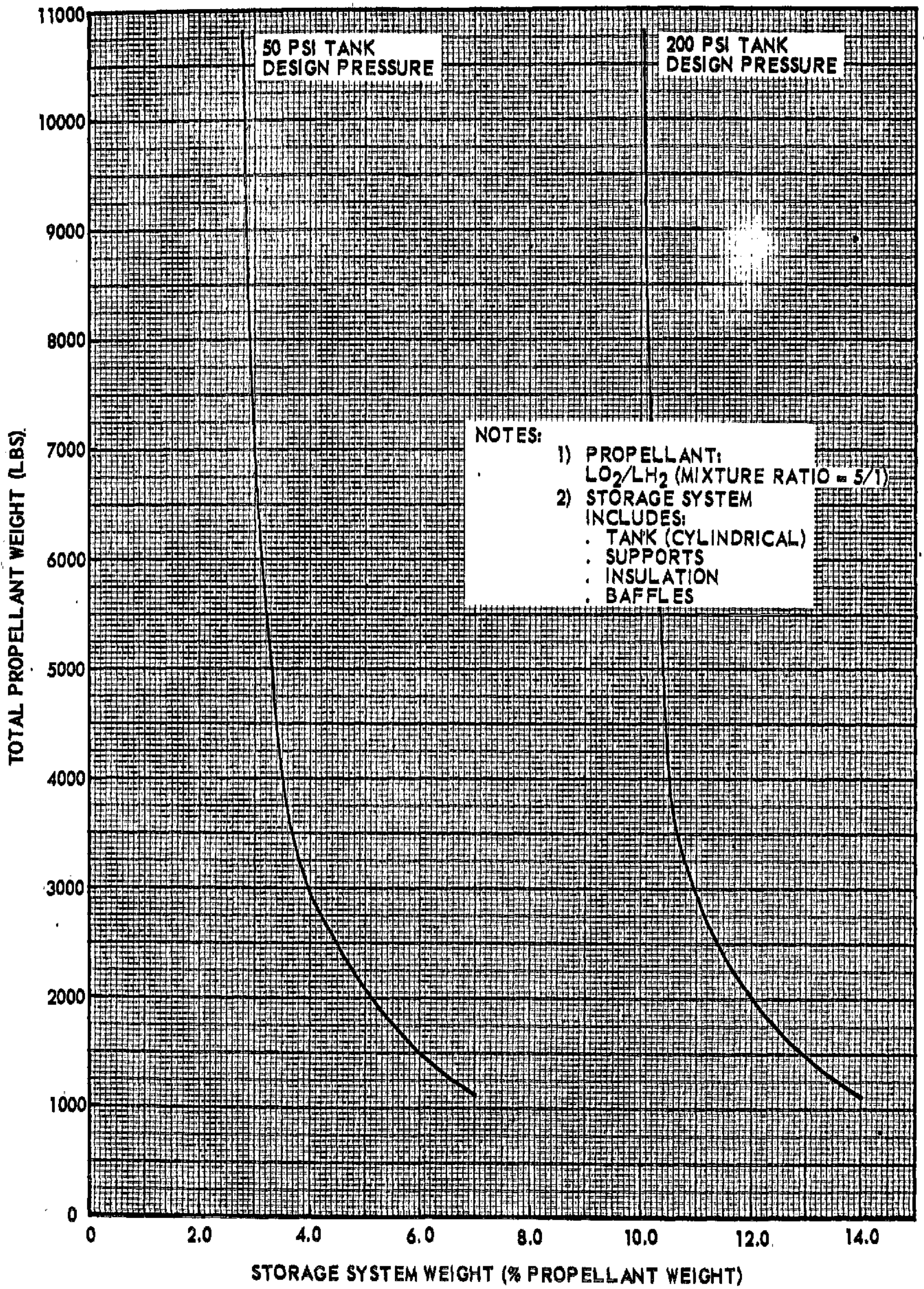


Figure 6C-2 CRYOGENIC PROPELLANT STORAGE SYSTEM WEIGHTS maximum translunar flight duration is estimated to be 80 hours. The approximate total storage times for the recommended LEM mission capabilities are shown in the table below.

	1 Day (Cold Side)	2 Days (Hot Side)	7 Days (Cold Side)
Descent Stage Tanks	80	80	80
Ascent Stage Tanks	104	128	240
Reaction Control Tanks	104	128	240

The two critical environment conditions which occur during these storage periods are the maximum heating and the maximum cooling conditions. The maximum heating conditions occur during either translunar flight with the tanks solar oriented or lunar rest on the hot side of the moon. The maximum cooling conditions occur during either the translunar flight with the tanks oriented to deep space or on the cold side of the lunar surface. The heat input during earth launch is of short duration and the tanks are protected during this period by the C-5 fairing. Heat inputs from the LEM exhaust plume and through conduction paths to the engine were estimated and found to be insignificant due to the short exposure times.

The LEM tankage requires thermal protection in order to maintain the propellant temperatures within allowable limits during storage. Storage of liquid hydrogen and liquid oxygen in the space environment poses the problem of reducing the heat leakage into the tankage. With these propellants, low heating rates are necessary to prevent excessively low propellant density and to maintain low propellant vapor pressure. Storage of propellants such as nitrogen tetroxide and Aerozine-50 poses the dual problem of maintaining sufficiently low temperatures to prevent excessive propellant vapor pressures as well as maintaining sufficiently high temperatures to prevent propellant freezing. The storage limits of these propellants are shown in the following table.

Propellant	Storage Temperatures
N_2O_4	40 to 120^oF
Aerozine-50	40 to 120^oF
Liquid H_2	-425 to -415^oF
Liquid O_2	-298 to -272^oF

The analysis of thermal protection methods for these two propellant combinations are summarized separately in the following paragraphs.

Storable Propellants

The only two thermal protection methods considered for storable propellants during this study were: (a) thermal control coatings and (b) insulation. Evaluation of these methods is accomplished for the fuel (Aerozine-50) and the conclusions are then applied to the oxidizer (N_2O_4). This is done since the thermal properties of N_2O_4 are sufficiently similar to those of Aerozine-50 to make the evaluation representative of both propellants.

Figure 6C-3 presents a temperature-time history of gold coated, non-insulated, Aerozine-50 propellant tanks for the extreme cold condition. From Figure 6C-3 it can be seen that pre-conditioning to 70°F prior to earth launch and gold coating are sufficient protection for the descent propulsion tanks (approximately 2500 lbs. of Aerozine-50 per tank and approximately 5000 lbs. of N_2O_4 per tank). Due to the large thermal mass in these tanks the gold coating is more than adequate to maintain the propellant temperature drop within limits for the 80-hour translunar flight. The ascent tanks will require approximately 850 lbs. of Aerozine-50 per tank and 1700 lbs. of N_2O_4 per tank. Figure 6C-3 shows that for a 24-hour lunar cold side mission, pre-conditioning to 70°F and gold coating are not satisfactory for these tanks. Therefore, some insulation is required. The reaction control tanks contain a relatively small mass of propellant (115 lbs. of Aerozine-50 and 230 lbs. of N_2O_4) and will also require insulation.

Figure 6C-4 shows the propellant temperatures after 240 hours of exposure to the extreme cold condition for gold coated tanks with various thicknesses of multi-layer radiation shield insulation. The minimum thickness of this type insulation from an installation standpoint is approximately 0.10 inches. From Figure 6C-4, it is apparent that this minimum thickness of insulation is sufficient to maintain the reaction control tanks and the ascent propulsion tanks within limits for a 7-day lunar cold side mission.

Figure 6C-5 shows the propellant temperature increase for a gold coated non-insulated tank exposed to the condition of maximum heating during the translunar flight (i.e., tank oriented toward the sun). From this figure, it is apparent that the descent tank temperatures are satisfactory without insulation. Figure 6C-6 shows propellant temperature-time histories for gold coated tanks incorporating 0.1 inches of super-insulation exposed to the maximum heating during the translunar flight and to the lunar hot side environment. This curve shows that 0.1 inches of super-insulation is sufficient for maintaining the temperature of the ascent and reaction control propellant tanks within limits for a 2-day lunar hot side mission.

In view of the above analysis the following thermal protection is recommended for the LEM storable propellant tanks:

Descent Tanks	Gold Coating Only
Ascent and Reaction Control Tanks	0.1 inches of super-insulation with a low emissivity surface

The above thermal protection will maintain suitable temperatures for storable propellants during the recommended LEM mission. Since the maximum conditions used in the above analyses are not expected to occur on any one mission, the recommended thermal protection methods are conservative.

Cryogenic Propellants

The extremely low temperatures of cryogenic propellants makes use of insulated tanks mandatory. In addition, heat leakage through lines and supports must be minimized. In order to obtain the lowest propellant storage system weight, trade studies between the thermal protection weight and the total tankage weight are required. The method of approach and significant conclusions for these trade studies are summarized below.

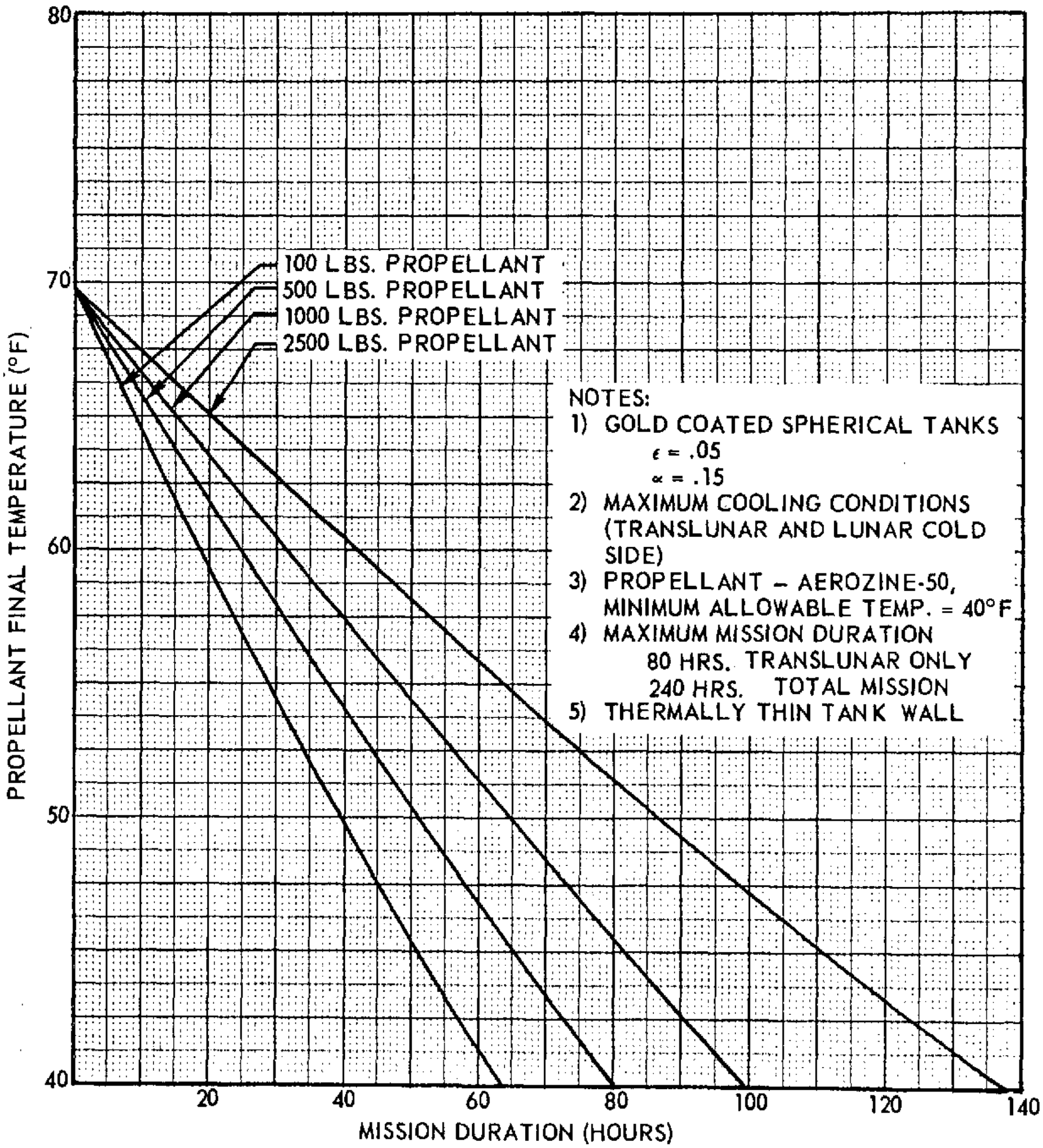


Figure 6C–3 STORABLE PROPELLANT COOLING RATE (NON-INSULATED TANKS)

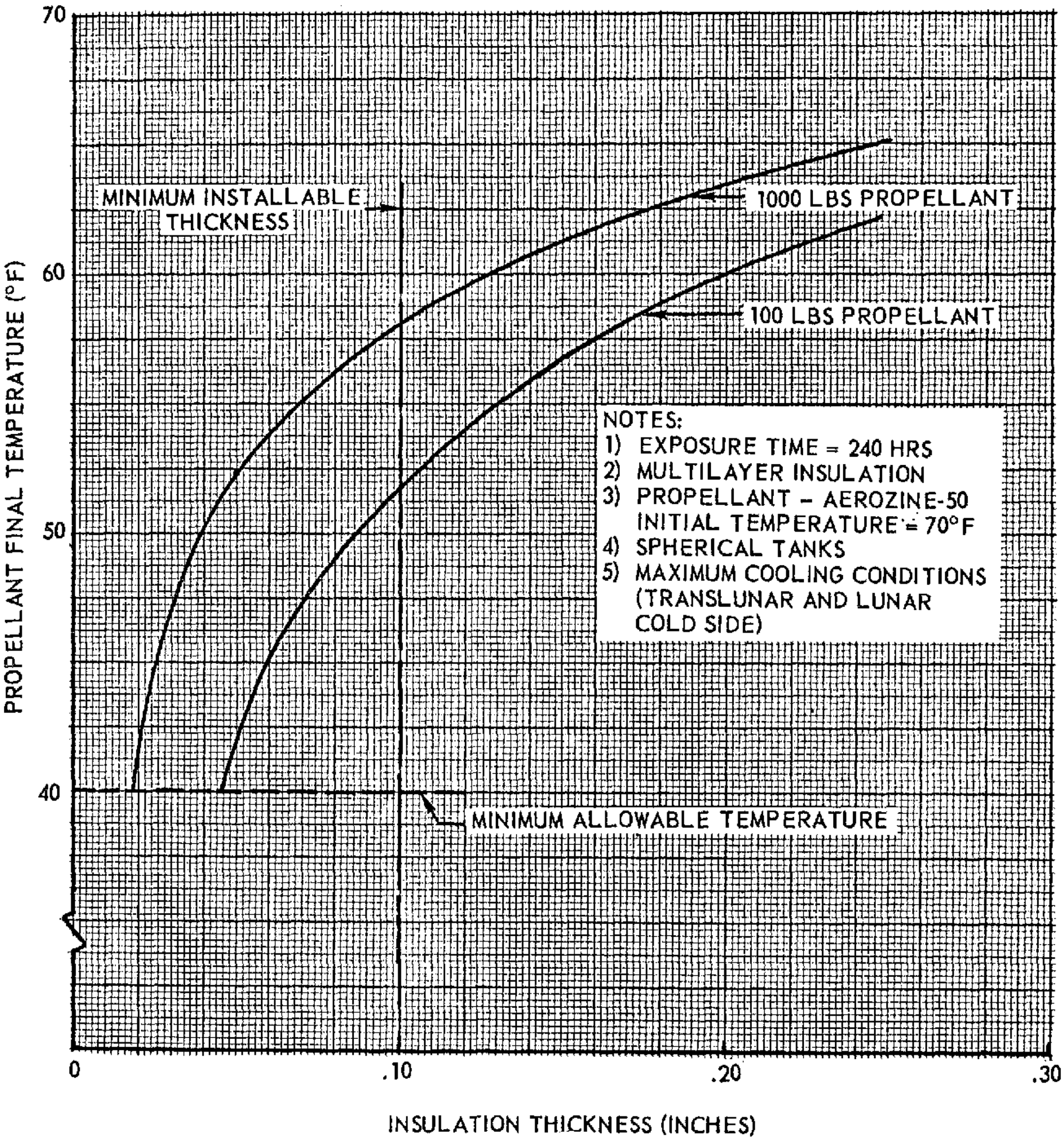


Figure 6C-4 **STORABLE PROPELLANT INSULATION REQUIREMENTS**

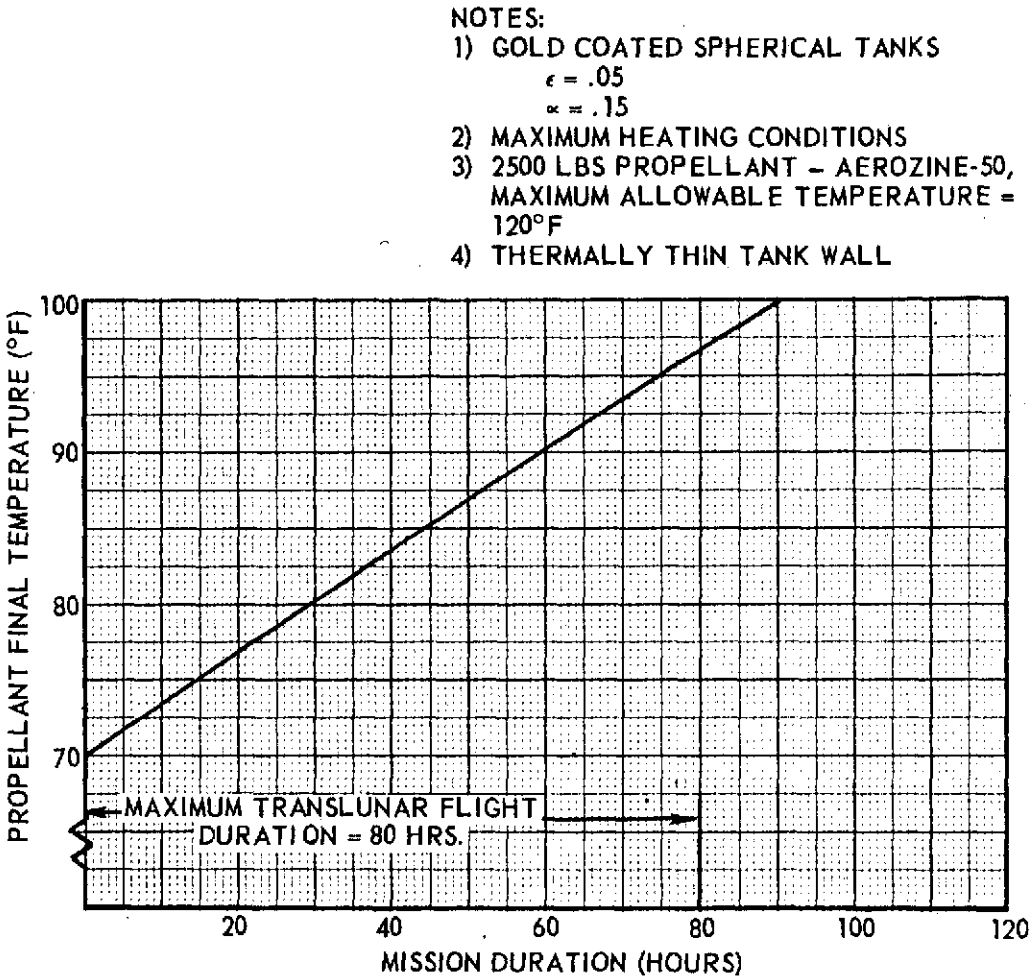


Figure 6C-5 STORABLE PROPELLANT HEATING RATE (NON-INSULATED TANKS)

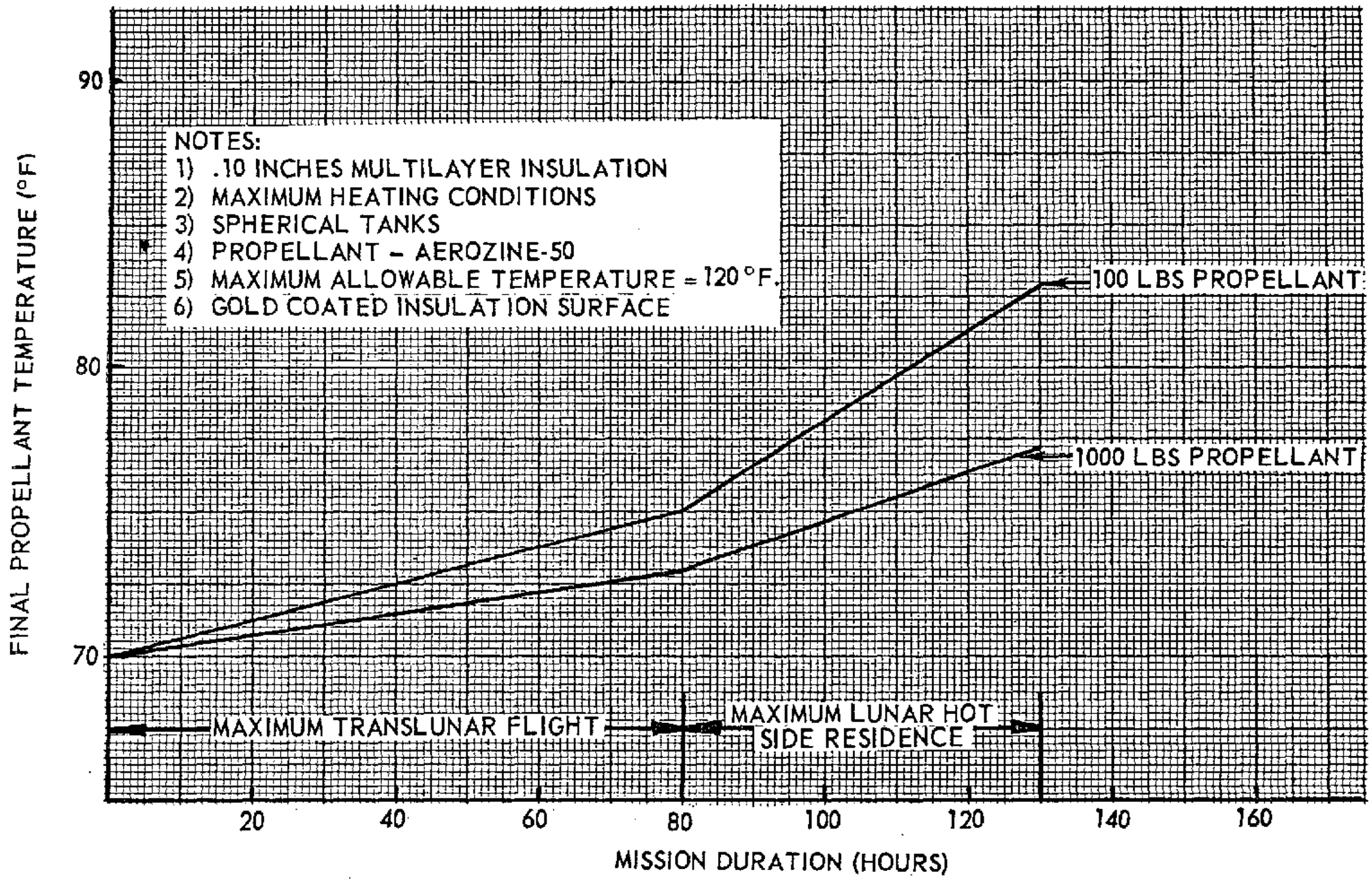


Figure 6C-6 STORABLE PROPELLANT HEATING RATE (INSULATED TANKS)

The first step in the cryogenic propellant thermal protection trade studies was to determine the range of tank heat inputs for a typical LEM mission. The primary heat paths considered were propellant lines, supports and walls. This investigation showed that the major heat leakage to the tanks is through the insulated walls.

The tank weight increase for the above range of heat inputs was then established. This increase occurs because of the higher propellant vapor pressures and lower propellant densities resulting from the increased propellant temperatures. Next, a design point for mission heat input was determined. This design point corresponds to the lowest combination of tank and insulation weight. The tank was considered to be non-vented (i.e., no boil-off). Use of a vented system for the relatively short LEM mission duration was found to be undesirable from a weight standpoint.

Finally the thickness of multi-layer radiation shield insulation was determined for the selected heat input design point. This thickness was then doubled to account for the potential effects of stratification. Figure 6C-7 presents the resulting storage system weight (tank support and insulation as a function of propellant quantity).

As a result of the above analysis the recommended thermal control system for LEM cryogenic propellants is multi-layer radiation shield insulation and a non-vented tank.

4.0 PROPELLANT TANK PRESSURIZATION

The pressurization system provides the driving force required to transport the propellants from the storage tanks to the engine pumps for pump fed engines and to the engine combustion chamber for pressure fed engines. For the LEM, the pressurization system must include provisions to accomplish the following specific requirements:

(a) Provide a relatively constant feed pressure for approximately 700 to 1000 seconds of engine operation
(b) Provide capability for 2 to 5 engine starts
(c) Provide provisions for engine throttling

The following types of pressurization systems were considered for meeting the above LEM requirements:

(a) High pressure stored gas
(b) Hot gas generator
(c) Vaporized propellant
(d) Main tank injection

The first step in evaluating these systems was to eliminate the complex methods requiring extensive development. On this basis the hot gas generator and main tank injection methods were removed from consideration. Hot gas generators are relatively complex, requiring, in effect, a small throttleable liquid or solid propulsion system capable of multiple restarts and a system for cooling and filtering the exhaust gases prior to injection into the propellant tanks. The method of main tank injection involves

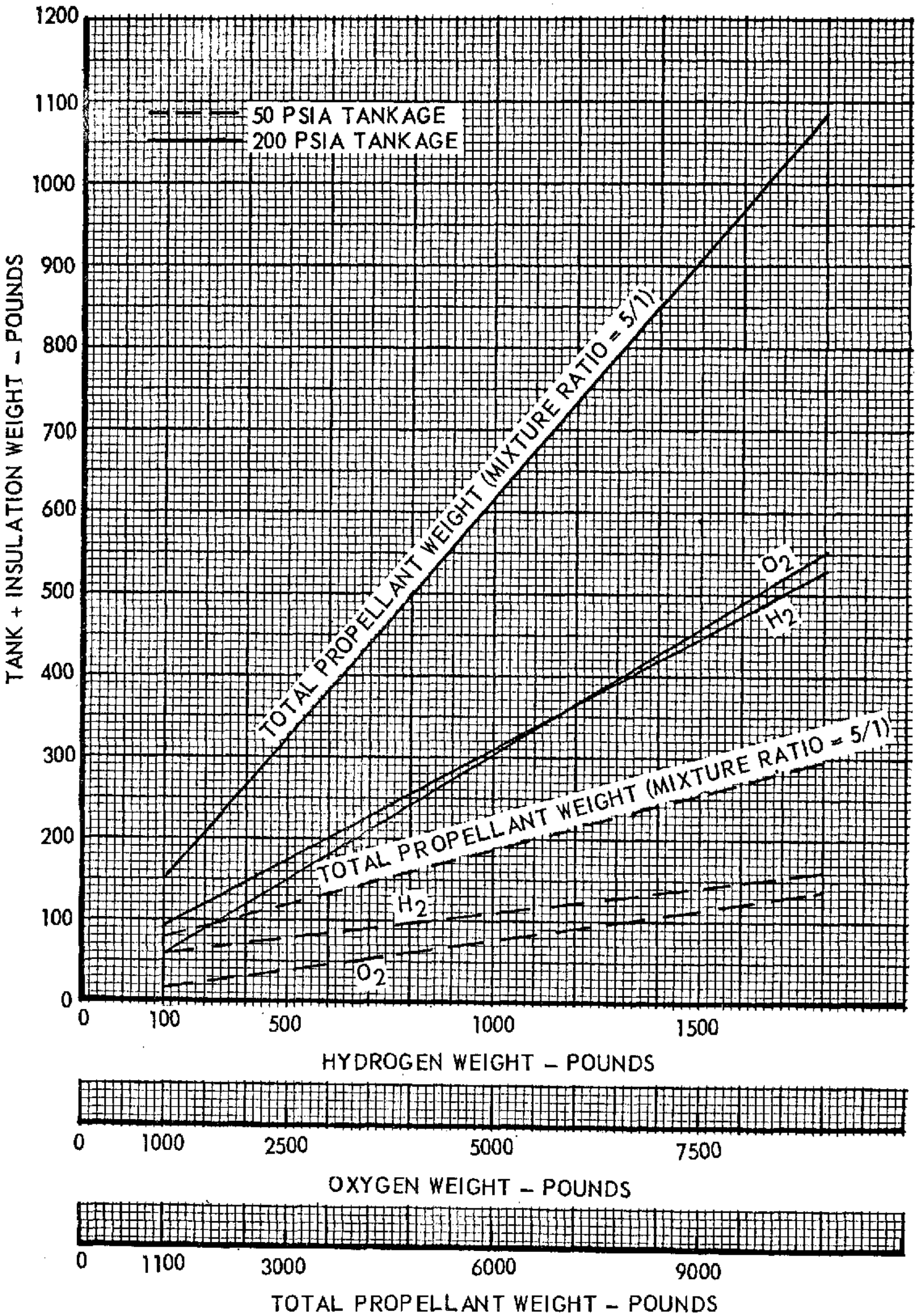


Figure 6C-7 TOTAL TANK AND INSULATION WEIGHT VS. CRYOGENICS PROPELLANT WEIGHT cross-feeding small quantities of hypergolic propellants (i.e., fuel into oxidizer tank and vice versa) in order to produce hot, high pressure combustion gases for pressurization. This method has not been developed sufficiently to prove actual feasibility or to establish performance.

The major factor affecting the feasibility of the vaporized propellant method is the required heating rates. For a 15, 000 lb thrust engine the required heating rate exceeds 100, 000 BTU/hr for either a cryogenic or a storable propellant system. For the LEM, the only feasible method to obtain such a high heating rate is to use a regeneratively cooled engine. A regeneratively cooled engine using storable propellants appears undesirable due to development problems. (Reference Paragraph 6.3). Regeneratively cooled cryogenic engines are being developed, however, the integration of this cooling method with the pressurization system has not been accomplished and would require considerable development. In addition, extensive plumbing problems and control complexity would result if this method were used with a multiple engine configuration. Therefore, the vaporized propellant method was eliminated due to complexity and development requirements.

High pressure stored gas pressurization systems have been used extensively and require almost no additional development. In addition, the stored gas method is relatively simple, provides high reliability, and meets all the requirements of the LEM propulsion system. The stored gas systems presently in use employ nitrogen and helium as the pressurant. Since pressurization of the propellant tanks requires a given volume of gas at a given pressure, the storage volumes of the two gases are the same (neglecting complexibility effects and assuming the storage pressures are the same). Therefore, the only significant difference in weight between the two systems is the weight of the required quantity of gas. Since nitrogen is approximately 7 times more dense than helium, the helium system was selected.

A brief investigation was conducted to determine the advantages obtained by heating the helium gas. For a 15, 000 pound thrust engine, a heating rate of approximately 100, 000 BTU/per hour is required in order to provide 200°F helium at the propellant tanks. However, as previously noted, there appears to be no suitable method to supply heat at this rate. Therefore, the system that best satisfies the LEM requirements is an unheated helium system employing high pressure storage.

Figures 6C-8 and 6C-9 show the estimated weight of this type system for cryogenic and storable propellants, respectively. The curves are based on use of a 3000 psia spherical titanium storage tank. For comparison purposes, the weight of a heated helium system is also presented.

5.0 PROPELLANT STORAGE AND PRESSURIZATION SUMMARY CURVES

Figures 6C-10 and 6C-11 present the total weights of complete storage and pressurization systems for cryogenic and storable propellants, respectively. The weights include tanks, tank supports, thermal protection required and the pressurization system. The weight presented by these two curves were used to establish the LEM propulsion system weight.

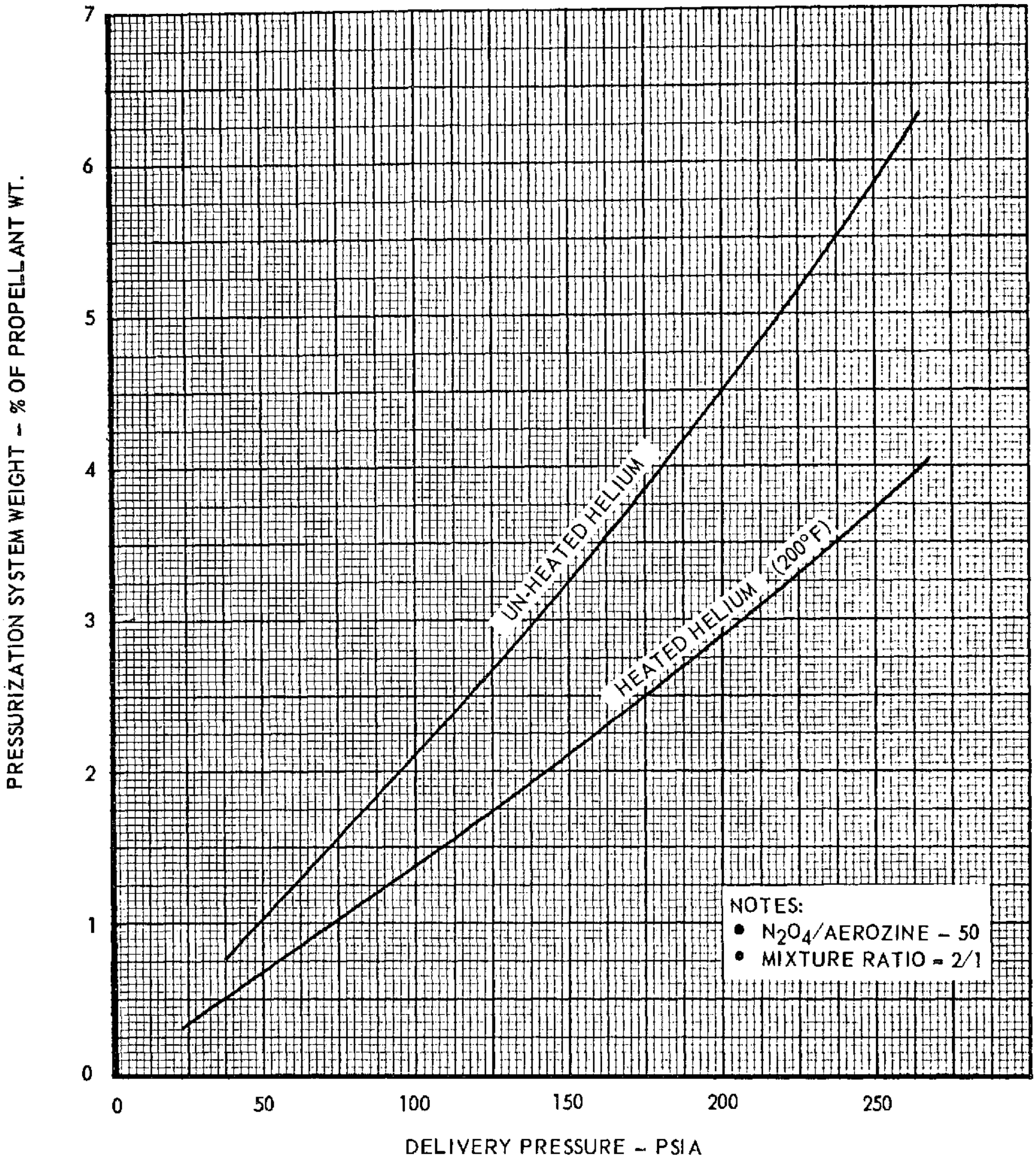


Figure 6C–8 STORABLE PROPELLANT PRESSURIZATION SYSTEM WEIGHT VS. DELIVERY PRESSURE

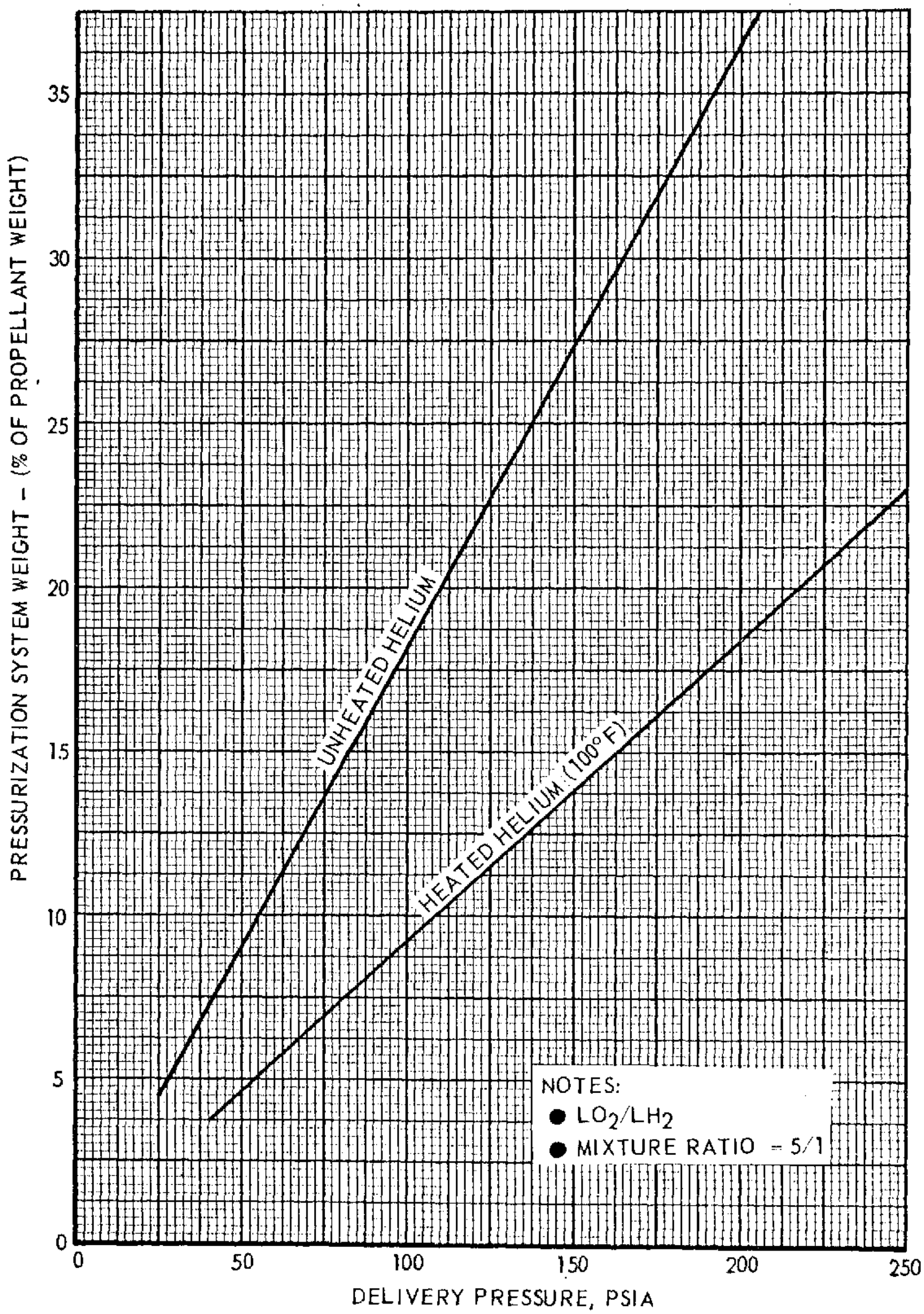


Figure 6C–9 CRYOGENIC PROPELLANT PRESSURIZATION SYSTEM WEIGHT VS DELIVERY PRESSURE

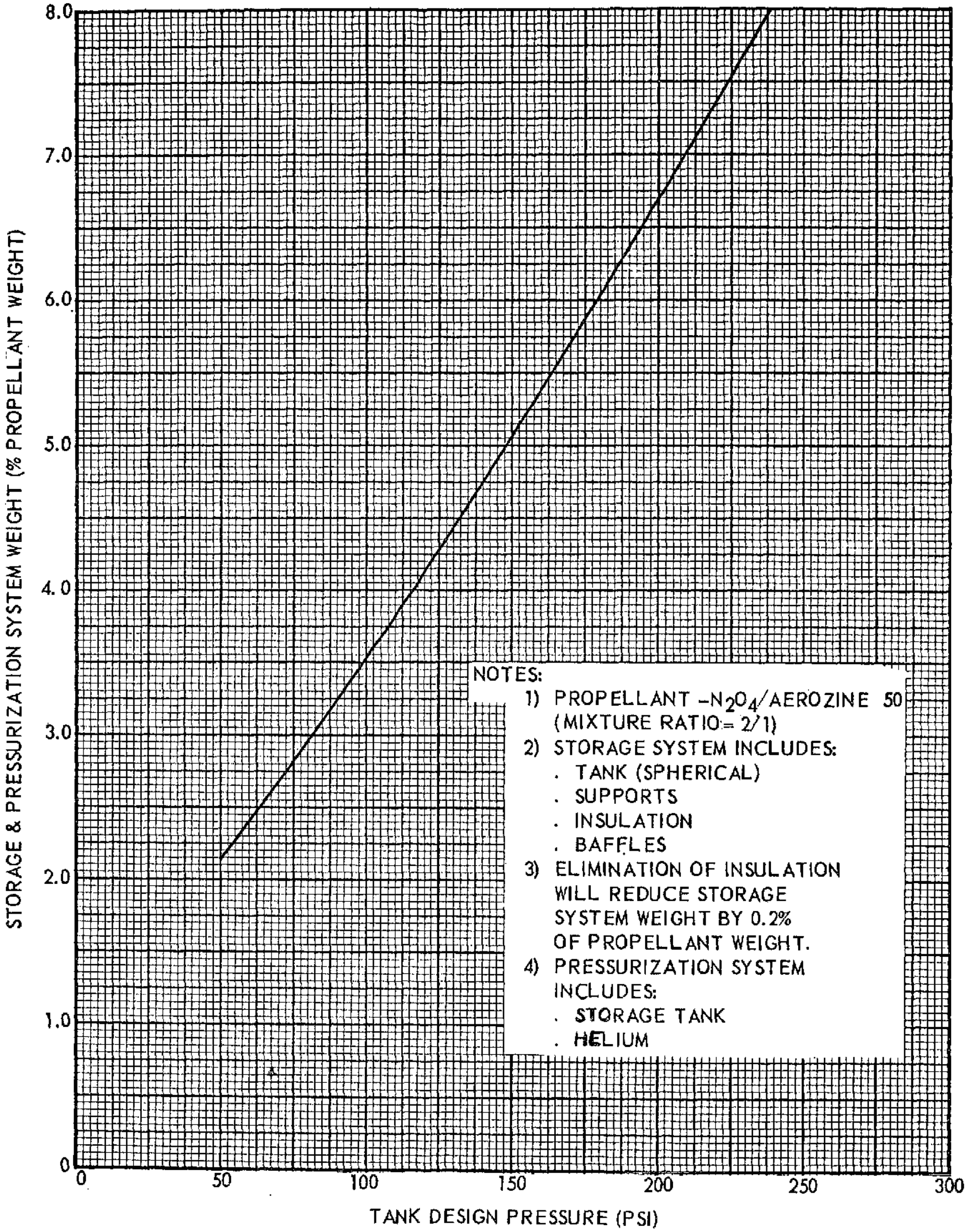


Figure 6C-10 STORABLE PROPELLANT STORAGE & PRESSURIZATION SYSTEM WEIGHTS

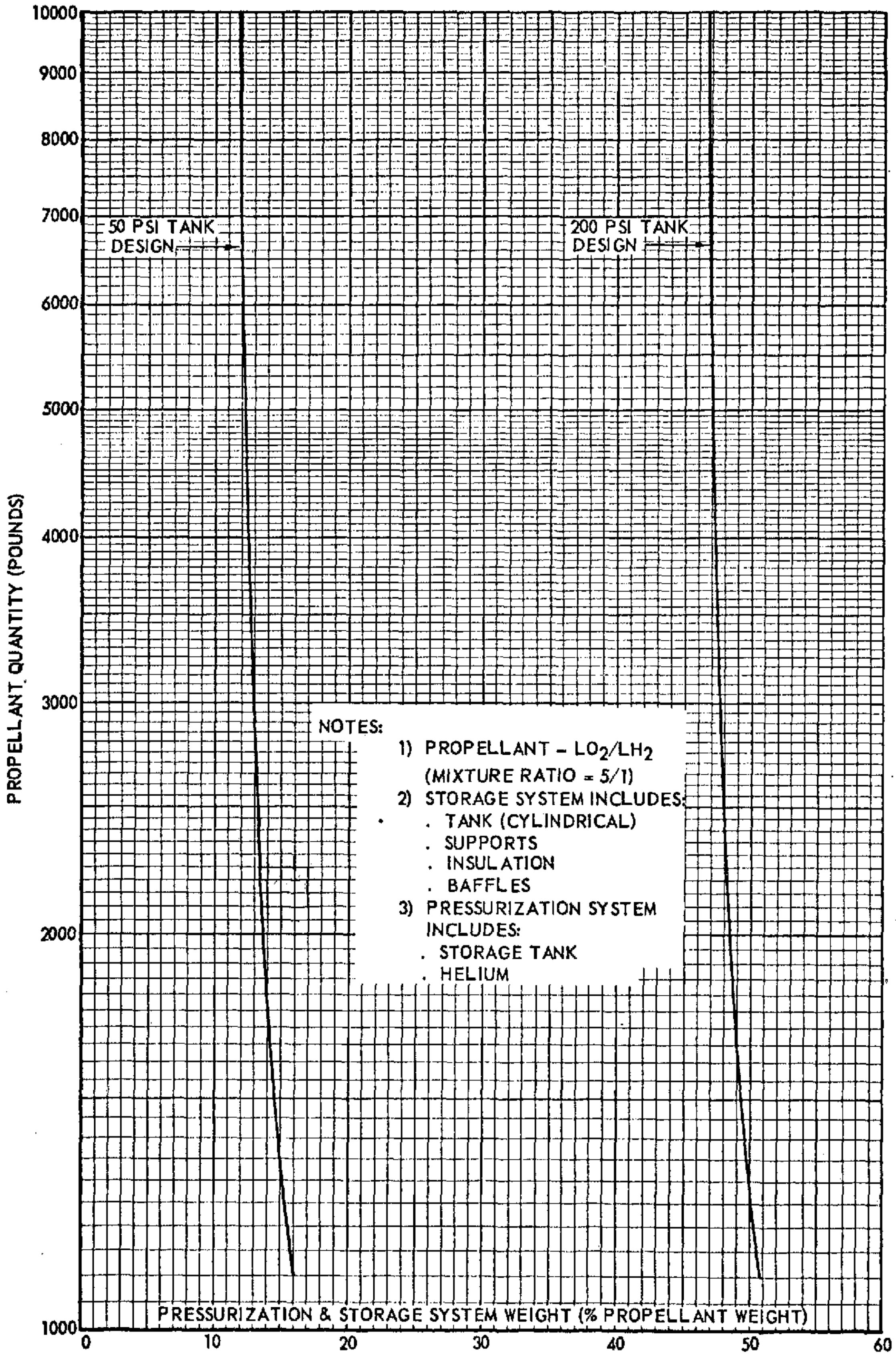


Figure 6C-11 CRYOGENIC PROPELLANT STORAGE AND PRESSURIZATION SYSTEM WEIGHTS

APPENDIX 6D

ROCKET ENGINE COMPONENTS

1.0 GENERAL

The purpose of this appendix is to define the development status of the major components of the engine system and to indicate their influence in the selection of the propulsion system concept.

2.0 ENGINE FEED SYSTEM

Both pump feed and pressure feed propellant systems were considered during this study. The considerations given to each are presented in the following paragraphs.

(a) Pump Feed System - The major advantage of a pump feed system is the ability to operate at high chamber pressures. This results in a decrease in thrust chamber size and weight. In addition, a pump feed system allows a lower tank pressure which reduces the propellant system weight. The above weight savings must be balanced against the weight of the pump. The disadvantages of a pump feed system are:

(1) System complexity
(2) Reliability associated with rotating machinery
(3) Disposal or handling of residual propellant during engine off periods
(4) Slow response time for start and throttling operations
(5) Pump cool down problem (cryogenic propellants)

The concensus of manufacturers indicated that development of a new pump feed engine system requires a minimum of 24 months to acquire preliminary flight rating. Therefore, it was necessary to limit consideration of this type system to existing developments. For earth storable propellants, no existing developments were found which could be modified to provide for throttling. Current development of pump feed systems with cryogenic propellants is limited to the RL-10 engine.

(b) Pressure Feed System - A pressure feed system is considered to be more reliable than a pump feed system due to simplicity. Also, the development time for a pressure feed system is considerably reduced over a pump feed system. The consensus of engine manufacturers is that the development time for a pressure feed system will be less than 18 months. The exact development time depends on the chamber cooling method employed. The lesser development time for pressure feed systems is realized because of the less critical temperature control, simplified starting techniques, simplified control system, and minimized residual propellant handling problems. However, the pressure feed system penalizes the over-all propulsion system due to its larger engine size and its higher over-all weight.

3.0 CHAMBER COOLING METHODS

Three general methods of chamber cooling are: radiative, regenerative, and ablative. In addition, combinations of the above methods require consideration due to the apparent weight saving to be realized. These combinations consist of ablative or regenerative cooling of combustion chambers with radiatively cooled skirts.

A. Radiation Cooling

Radiation cooling has been employed on small rocket engines operating intermittently. Recent tests have been conducted on larger chambers with full radiation cooling. However, material problems have limited successful operation to engines with low chamber pressures. These engines are not considered to be sufficiently developed for use on the LEM.

Radiation cooled skirts have been proposed for employment on ablative chambers. Initially, this combination appeared to offer a weight saving. However, detailed evaluation shows that the weight of the connection flanges will probably offset much of the weight savings in the skirt. In addition, the problem of cooling the skirt in multiple installations (i.e., clustered skirts radiating to each other) is still to be solved. Serious vehicle installation problems must be solved in order to use engines with radiation cooled skirts on the LEM. These problems are: (1) radiative heating of other systems in the vehicle and (2) susceptibility to damage from debris, micrometeoroids, and operational accidents.

In view of these problems, it was concluded that radiation cooled nozzle skirts should not be considered for the LEM.

B. Regenerative Cooling

Regenerative cooling has more development background than any other rocket engine cooling method. All current liquid propellant engines employ this method of cooling. Design and quality control techniques have been thoroughly developed for this cooling method. Radiant heat and chamber size are minimized with regeneratively cooled engines. This cooling method provides an engine with a characteristic long life due to the low operating temperature of the materials.

(1) Cryogenic Application

A cryogenic thrust chamber can be adequately cooled by the regenerative method using the high heat capacity of the hydrogen fuel. This type of chamber cooling for a cryogenic engine is considered acceptable for the LEM. However, the development time for a new design would be 18 to 24 months which would be marginal in meeting the LEM schedule. The long time for development is associated with the lead time required to design and fabricate the chambers. Thus, it was considered advisable to limit the consideration of this type chamber to existing developments.

(2) Earth Storable Application

Many successful applications of regenerative cooling have been accomplished with earth storable propellants. However, the cooling capacity of the propellants has always been a development problem particularly with small engine sizes. The oxidizers, which have been compounds of nitric acids or nitrous oxides, have not been successfully used as coolants because of low heat capacity and low vapor pressures. The fuel is acceptable as a coolant, but is marginal in capacity because of the 2 to 1 mixture ratios or better which are necessary for acceptable performance. The limited capacity of the fuel has lead to delicate design problems and high cooling jacket and chamber pressures in order to eliminate coking and sludging in the coolant passages and to avoid vaporization in the injectors. Considering the throttling required in the LOR mission it is felt that a solution to the cooling problem would require significantly more time than is available for development.

A novel approach was reviewed which employed a combination of cooling techniques now being developed for the Surveyor vehicle. The concept is loosely referred to as 'regenerative cooling' but actually uses the regenerative principle only to cool the combustion chamber and assist with nozzle throat cooling. The nozzle throat is made of a high temperature ceramic. The nozzle beyond the throat employs radiation cooling. While this approach is of interest, it presents several problems when considered for the LEM application. These are:

(a) Multiple starts resulting in heating and cooling of the throat.

(b) Jacket purging and propellant freezing between operating cycles.

(c) Vehicle design as affected by the high radiant heat from the nozzle.

C. Ablation Cooling

Ablation cooled engines are relatively new in the propulsion field. However, they offer the unique advantages of a simplified chamber with a minimum of external radiant heat. Ablative chambers have been operated in experimental facilities satisfactorily for periods up to 400 seconds at thrust levels of 5000 to 8000 lbs. at sea level. Several such engines are under current development. At least one engine in the 2000 lb. thrust class is nearing completion of its Pre-Flight Rating Test. Several other engine programs, such as the Apollo Service Module engine and the Gemini and Apollo Reaction Control, will significantly improve the state of the art relative to ablation cooled engines. The ablation rates are sensitive to heat flux and flow characteristics and therefore dependent upon injector and chamber design. Although inherently heavy, these ablation cooled engines are of very simple, rugged design, and are the least susceptible to external damage. Therefore, ablative cooled engines are recommended for the LEM.

4.0 ENGINE INJECTOR TECHNIQUE

Both fixed and variable area injectors were considered in this study. Consideration was given on the basis of anticipated throttling requirements for the main propulsion system up to approximately 10 to 1. Because throttling with both injector types involves dynamic components, detailed evaluations of both design approaches were accomplished.

The fixed area injector has been used on all liquid propellant rocket engines developed to date. Throttling has been accomplished with such an injector by varying the flow rate of either or both propellants with upstream throttle valves. However, throttling for the LEM mission requires thrust changes up to 10 to 1 depending on the vehicle thrust to weight ratio and the number of thrust chambers utilized.

The problem associated with high throttling ratios using a fixed area injector and storable propellants is that the injector pressure drop varies as the square of the flow rate (or the throttle ratio). The maximum value in each case depends on the minimum injector pressure drop attainable. However, with a practical minimum injector drop the required feed pressure for high throttle ratios becomes quite high with pressure fed systems. This results in large tank and pressurant system weights. In addition, a difficult injector design problem exists in providing an acceptable spray pattern over a wide range of injector pressure drops.

A practical throttle ratio on the order of 4 to 1 has been demonstrated with fixed area injectors. It has been estimated that a reasonable maximum would be about 7 to 1. The minimum practical pressure drop across the injector has been determined to be about 6 psia. With this low injector drop a minimum chamber pressure has been determined experimentally in the range of 20 psia to support stable combustion. Stable burning is a function of propellant vapor pressure and can only be achieved when the total of injector pressure drop and chamber pressure are greater than propellant vapor pressure. In all instances in the study, a safety factor of 2 psia was used in establishing the minimum feed pressure for any particular propellant temperature. An example development of the application of fixed area injectors using storable propellants is the Surveyor motor currently under development with a throttle range of 3 to 1.

The technique of throttling cryogenic propellants with a fixed injector is not as difficult as with earth storables provided the liquid hydrogen is heated in a regenerative chamber. In this case, the hydrogen can be injected as a gas and the pressure drop square law does not apply. Therefore cryogenic propellants can be throttled through a fixed injector with relatively minor difficulty as has been demonstrated with the RL-10 engine. The problem of injecting unheated liquid oxygen and liquid hydrogen has not been explored.

Variable area injectors are limited in state of the art development and there is no flight experience to date. Almost every liquid rocket manufacturer has some type of development program in this area, but all readily agree that a probable 18 months experimental development effort would be required to provide a design suitable for use in an engine development program. Injector concepts vary from variable injector area arrangements, to sleeve pistons varying the number of fixed area injector holes to preheated gas injectors in which flow may be varied. In general, all appear to have one outstanding disadvantage, that of warpage of moving parts in close proximity to the extremely high temperatures in the injector region. There are several systems scheduled to go into conceptual evaluation tests during the summer and fall of 1962, but these systems are not adequately developed for consideration in the LEM Program.

5.0 GIMBALING

Engine gimbaling is used for thrust vector control on a majority of liquid propellant booster and sustainer engines being used in current space probes. The primary areas of concern relative to gimbal development are: bearing surfaces and the possibility of space welding, flexible lines for space application, actuators and power supply. Brief investigations into vacuum welding of the bearing surfaces indicate the problem can be resolved by using dissimilar materials and non-lubricated bearing surfaces. Dual actuators have been developed and are considered adequate for this application. It appears that flexible feed line problems can be solved with current design procedures.

The power for actuation was investigated with the resulting recommendation that a hydraulic system be used. Electrical power requires a large storage source for peak power loads and was therefore discarded in the study. Pneumatic actuation does not provide adequate control. Hydraulic power, on the other hand, provides maximum power requirements by use of accumulators and proportional control through the use of valves.

6.0 VALVES AND REGULATORS

Valves and regulators of the type required for the LEM propulsion system are in use in many flight systems at the present time. Although precise components may not be available as off-the-shelf items, space qualified components are considered feasible within the LEM development schedule.

APPENDIX 6E
SAMPLE CALCULATIONS FOR PROPULSION SYSTEMS EVALUATION

1.0 SAMPLE CALCULATION - PRELIMINARY PROPULSION SYSTEM STUDIES

1.1 General

The objective of this section of Appendix 6E is to illustrate the methods of analysis used in the generation of the data presented in Paragraph 6.3.1 (Figures 6-1 through 6-4). In general, the approach is a graphical method of problem solution with basic energy and performance equations used for development of the initial working curve. To illustrate the method of analysis a sample problem is shown based upon the following propulsion system and mission requirements.

(a) Propulsion System

- o Thrust level - 5000 lbs.
- o Single ablative thrust chamber
- o Pressure fed system
- o Propellant - Nitrogen Tetroxide - Aerozeine 50 (I_{sp} = 320)

(b) Mission Requirements

- o Descent velocity change - 6500 fps
- o Ascent velocity change - 6850 fps
- o Staging - landing gear and descent propellant tanks staged on the lunar surface

The basic propulsion system and mission requirements are discussed in more detail in Paragraph 6.3.1 with a detailed system description shown as System 1 in Table 6-III.

1.2 Method of Analysis

Using the following equations construct Figure 6E-1 for varying velocity change, system mass fractions and specific impulse:

(a) Characteristic velocity equation

$$\mu = {}^{\Delta V/I_{sp}\, g} = W_O/W_{BO}$$ (See Paragraph 3 for complete symbol list)

(b) Propulsion system performance equation

$$W_T/W_X = 1 - \mu \Big/ \left[\mu(1-MF) - 1\right]$$

where:
W_O = initial LEM weight
W_{BO} = LEM weight after application of a given ΔV
ΔV = ideal velocity change
I_{sp} = specific impulse

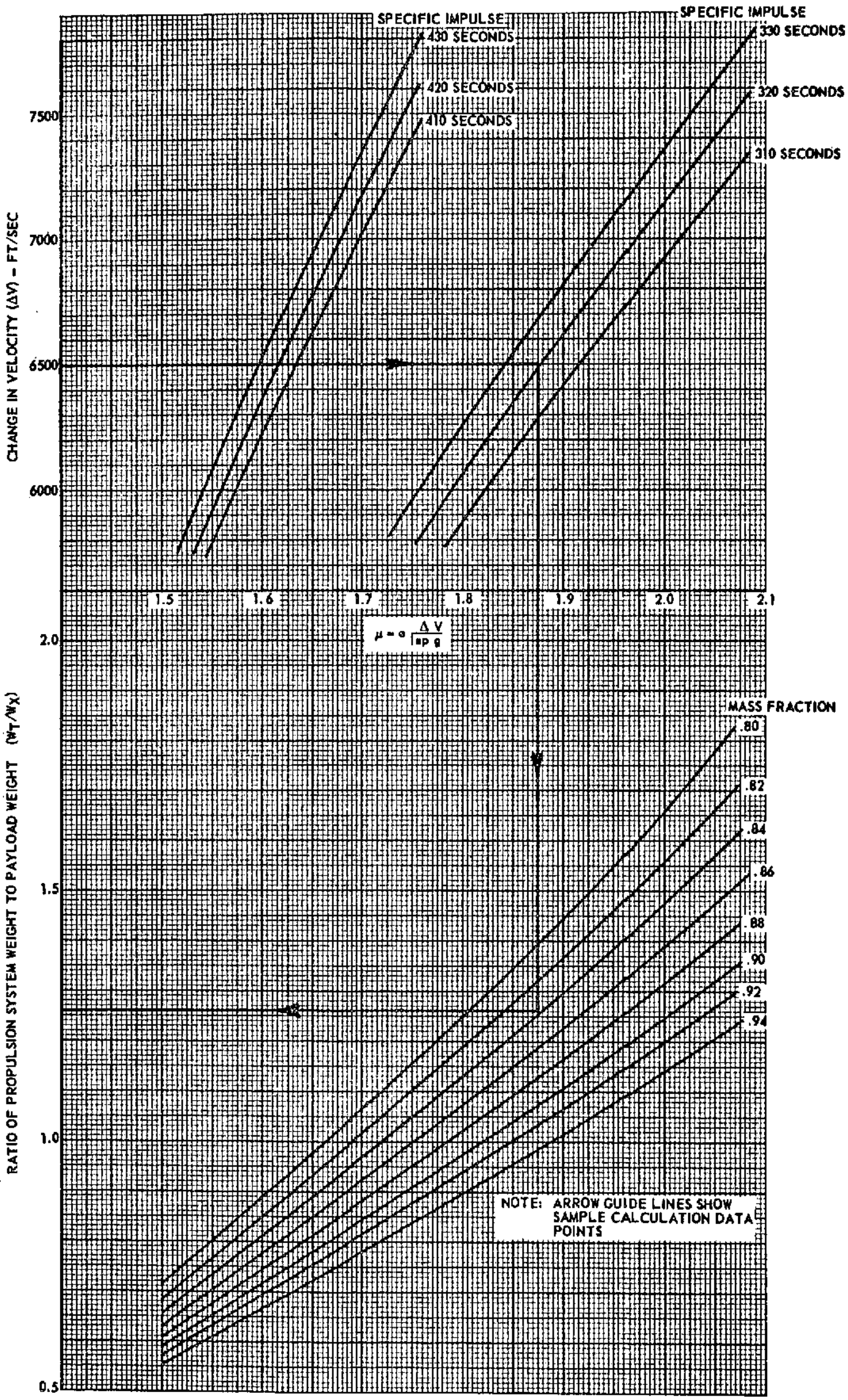


Figure 6E-1 PROPULSION SYSTEM PERFORMANCE

W_T = total propulsion system weight

W_X = stage payload weight

$$MF = \frac{\text{propellant weight}}{\text{propulsion system weight}} = \frac{W_P}{W_T}$$

The above equations relate the mission requirements to the type of propulsion system used in the vehicle. The arrows on Figure 6E-1 illustrate the point used for the first stage of the chosen sample problem.

Figure 6E-2 is constructed by selecting varying propellant weights and computing the mass fractions for the selected propulsion system at each propellant weight. The propulsion system weights (thrust chamber, accessories, propellant tankage, etc.) are discussed in detail and presented in parametric form as thrust dependent functions in Appendix 6B or as propellant weight functions in Appendix 6C.

Figure 6E-3 illustrates the data from Figure 6E-2 with stage payload weights superimposed on the curve. These payload weight lines are arrived at by selecting varying mass fractions for a particular stage and computing the corresponding propellant weights for the stage. The example noted in Figure 6E-1 by the arrows for an I_{sp} = 320 seconds and mass fraction of 0.84 gives a ratio of propulsion system weight to payload weight of 1.26. The corresponding propulsion system weight for a 10,000 lb. payload is (1.26)(10,000) or 12,600 lbs. The resulting propellant weight is (12,600)(0.84) or 10,600 lbs. Several calculations such as this for varying mass fractions and payload weights yields Figure 6E-3. Note that the point where the payload line intersects the mass fraction curve is the mass fraction, propellant weight and payload for the chosen 5,000 lb. thrust system.

Figure 6E-4 is generated by choosing varying payload weights from Figure 6E-3, and for the corresponding mass fractions and propellant weights computing the initial stage vehicle weights (initial LEM weight for Stage 1 and lunar launch weight for Stage 2). For example, for a first stage payload weight of 10,000 lbs., the mass fraction is 0.870 and the propellant weight is 10,100 lbs. The propulsion system weight is then 10,100/0.870 or 11,600 lbs. The initial LEM weight for the particular payload and propulsion system is 11,600 + 10,000 or 21,600 lbs. (Figure 6E-4). The same procedure will produce the curve consisting of second stage payload versus lunar launch weight.

Figure 6E-5 is derived from use of the previously generated curves. By selecting varying final stage payload weights and propellant weights from Figure 6E-3 and reading the corresponding initial LEM weight from Figure 6E-4 the lunar landed weight is the LEM weight minus the propellant weight. It follows that the lunar launch weight is the landed weight minus the staged weight (landing gear and tankage for the example here). The right hand portion of Figure 6E-4 yields the second stage payload (crew station weight) for the various lunar launch weights. Figure 6E-5 may then be plotted as crew station weight versus initial LEM weight.

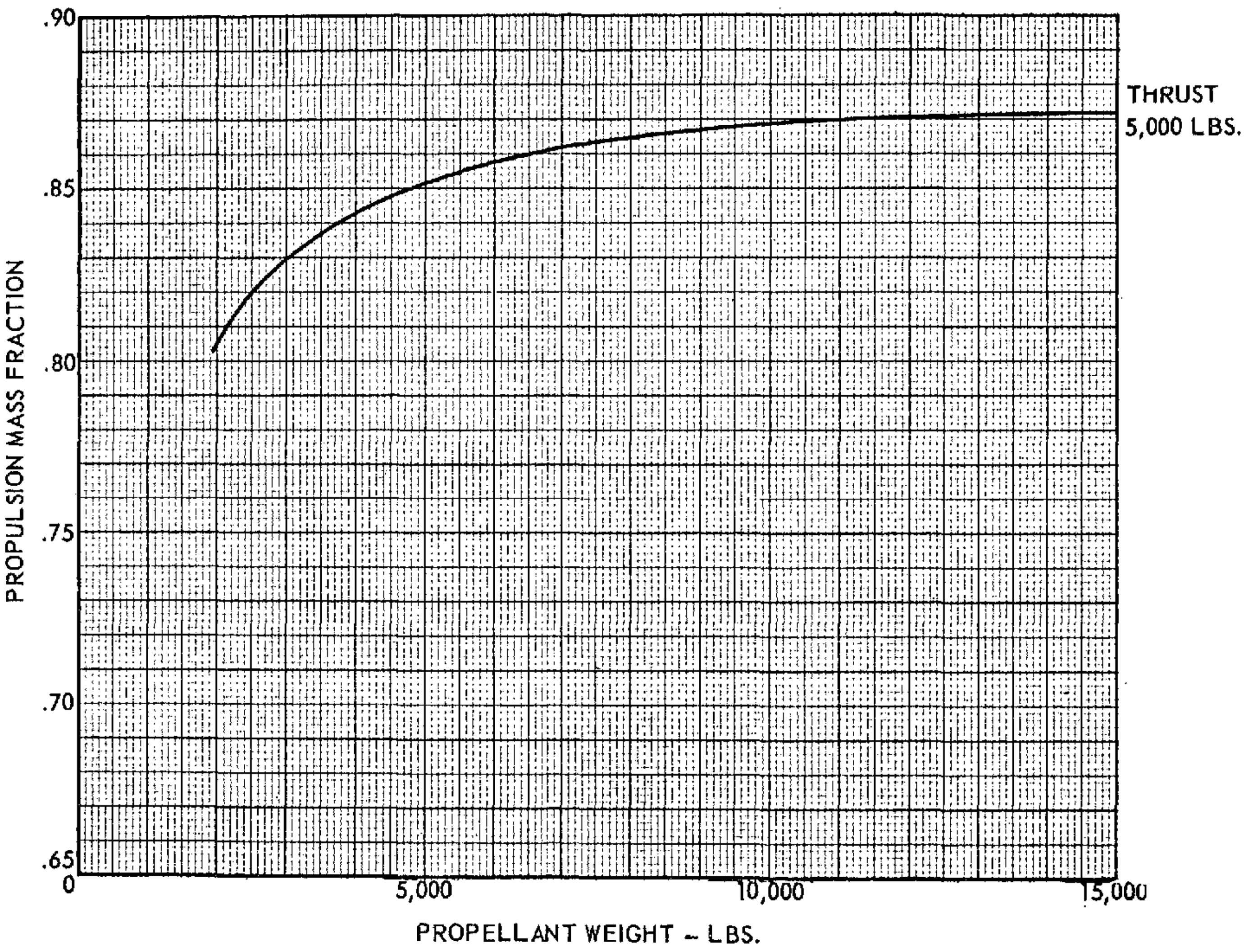


Figure 6E-2 PROPULSION MASS FRACTION

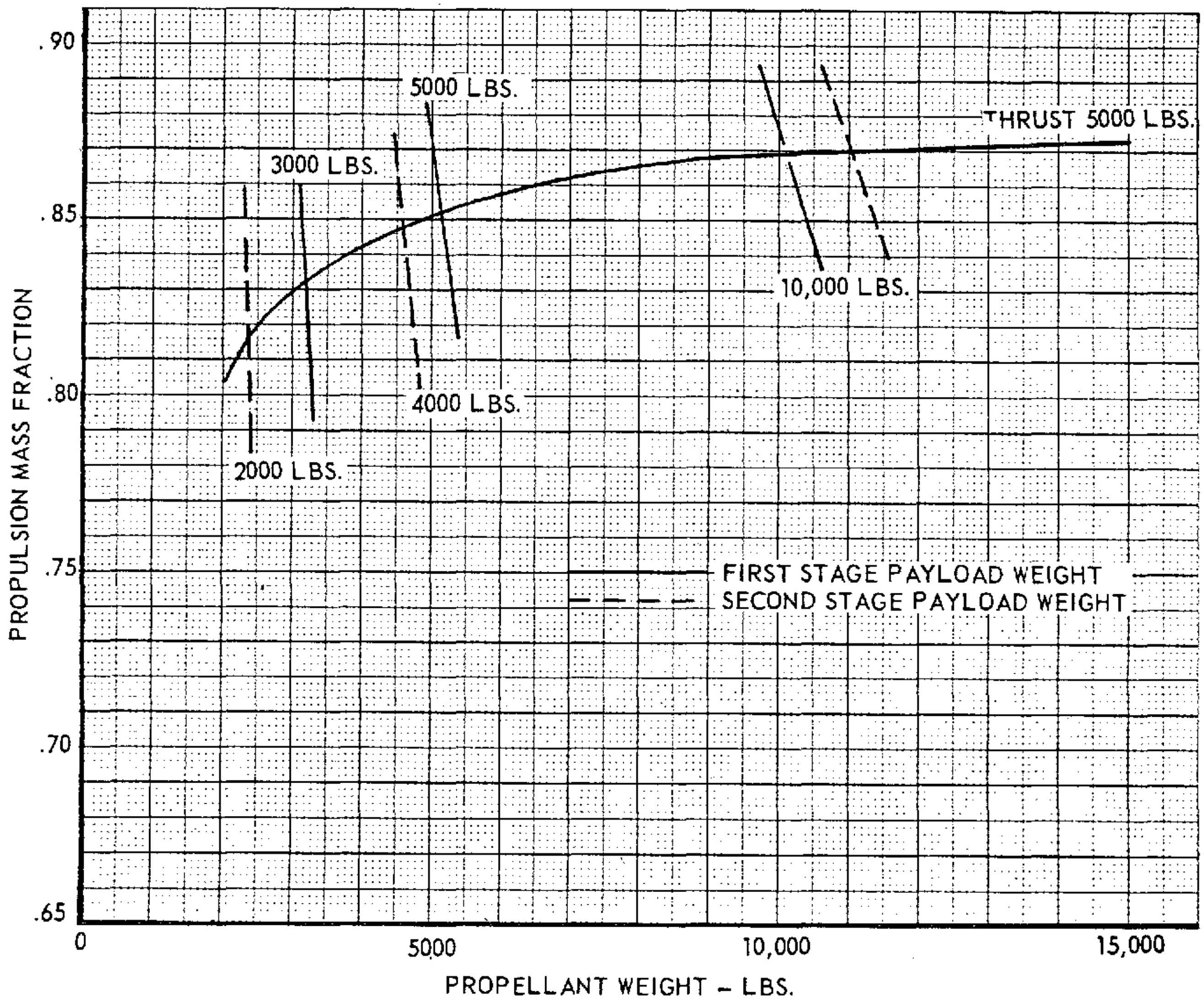


Figure 6E-3 PROPULSION MASS FRACTION – PAYLOAD WEIGHT RELATIONSHIP

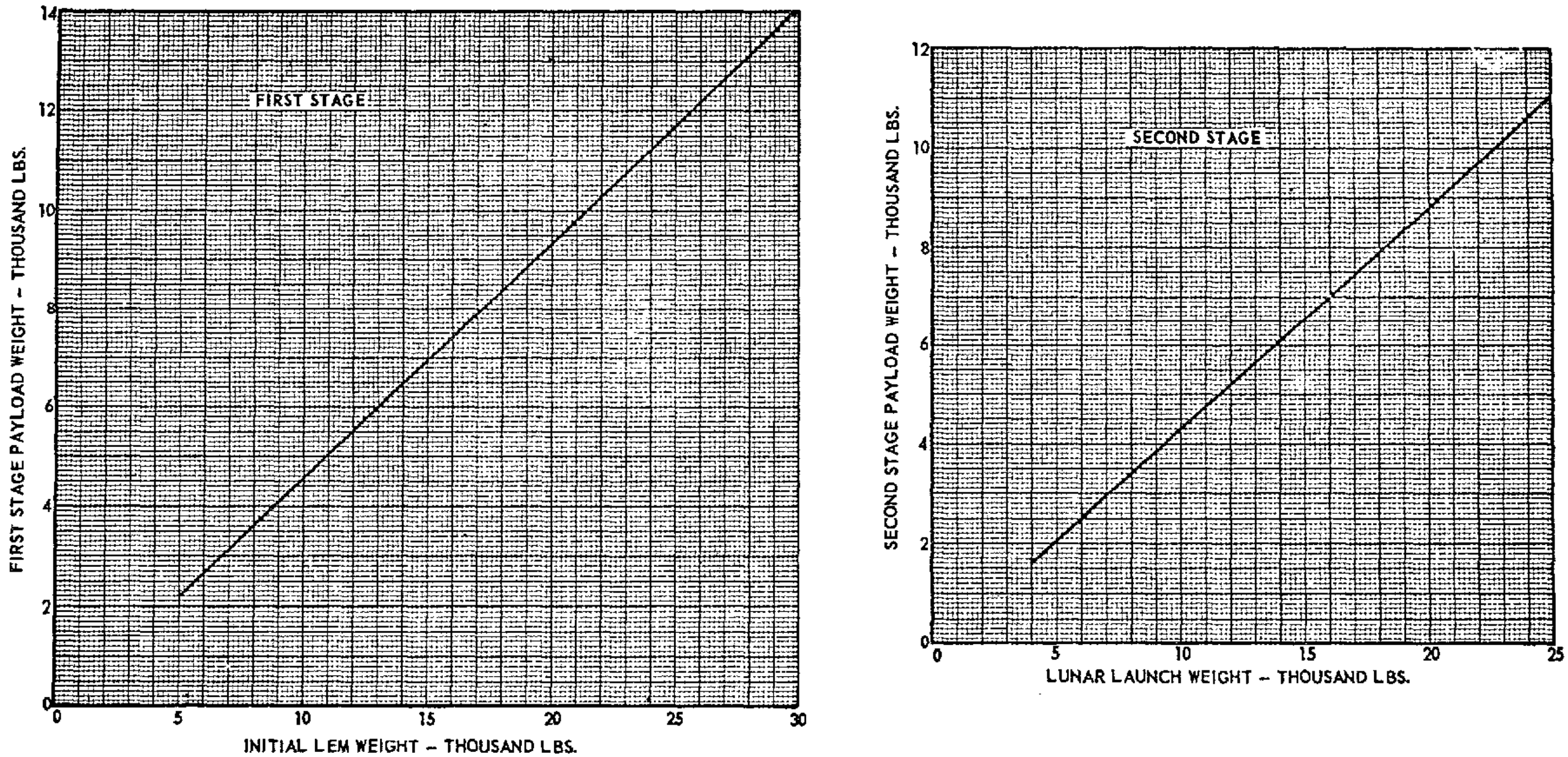


Figure 6E-4 STAGE PAYLOAD WEIGHTS

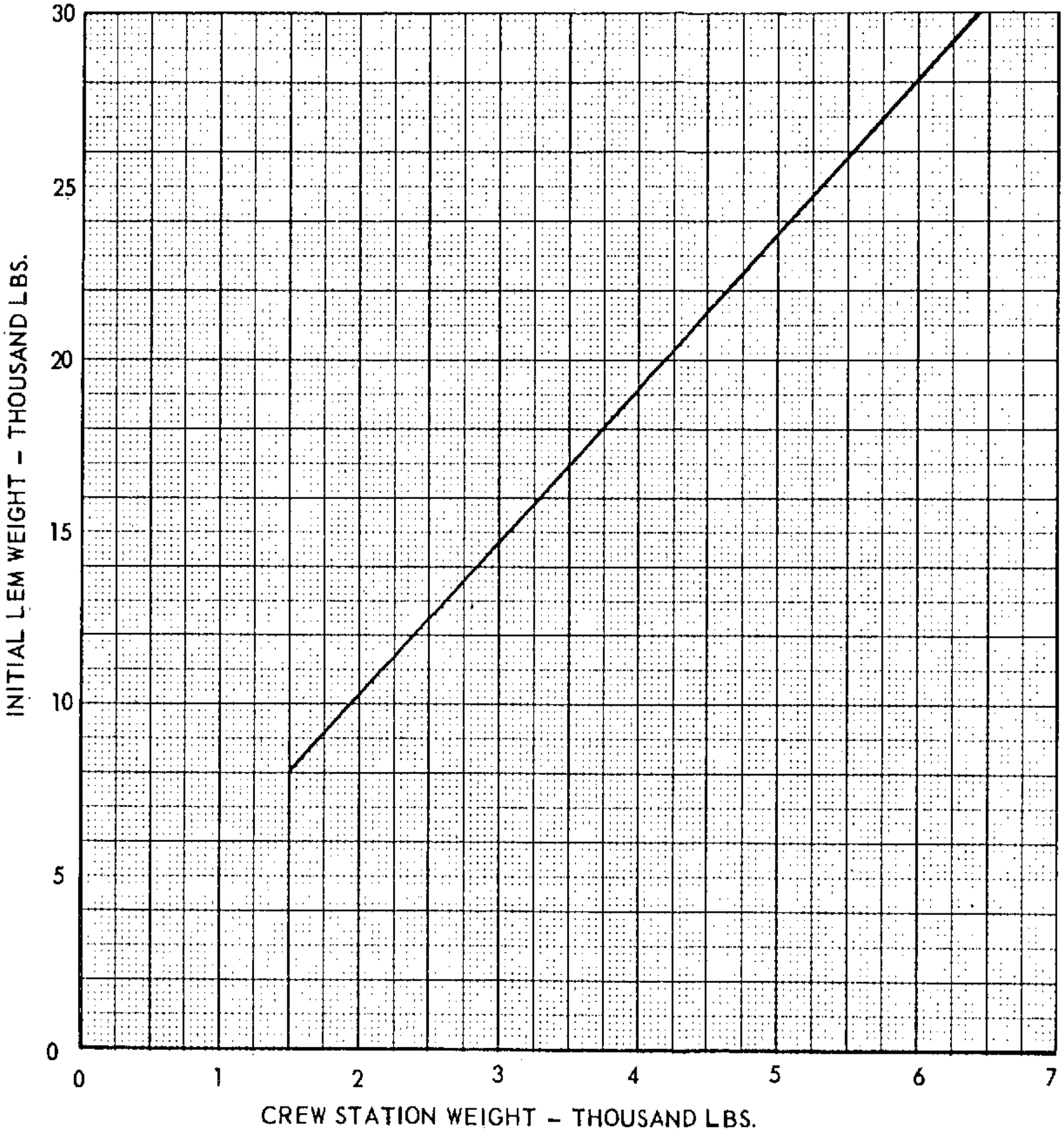


Figure 6E-5 VARIATION IN LEM WEIGHT WITH CREW STATION WEIGHT

For the example system chosen and a first stage payload of 10,000 lbs., the propellant weight is 10,100 lbs. (Figure 6E-3). From Figure 6E-4 the initial LEM weight is 21,600 lbs. The lunar landed weight is therefore 21,600 - 10,100 or 11,500 lbs. The landing gear weight is 575 lbs. for this particular landed weight and the propellant tankage is 357 lbs. By staging the gear and tanks the lunar lift off weight becomes 10,570 lbs. The crew station weight is 4,500 lbs. (Figure 6E-4). Several selections such as the one above were made to generate Figure 6E-5.

While the sample problem chosen for illustration was for a particular mission and propulsion system, many variations of mission, staging methods, and types of propulsion systems may be studied in parametric form by the above methods. The results presented in Paragraph 6.3.1 were obtained in this manner.

1.3 List of Symbols

μ	-	Ratio of LEM initial weight to final weight
g	-	Gravitational constant = 32.174 ft/sec^2
MF	-	Ratio of propellant weight to propulsion system weight
ΔV	-	Ideal velocity change
W_{BO}	-	LEM final weight after application of a given ΔV
W_o	-	LEM initial weight
W_p	-	Propellant weight
W_T	-	Total propulsion system weight
W_X	-	Stage payload weight

2.0 SAMPLE CALCULATIONS - FINAL PROPULSION SYSTEM PARAMETRIC STUDIES

2.1 General

Final propulsion system parametric studies were accomplished using the LEM gross weight as the basis of comparison. These studies differed from the preliminary evaluation studies in that:

(a) The crew station weight was approximately defined.
(b) A different reserve propellant philosophy was utilized.
(c) Parametric ascent and descent trajectory data were utilized.

Equations which describe the LEM weight at various points in the mission were derived to facilitate inclusion of the parametric trajectory data. The derivation of these equations is presented in the following paragraphs.

2.2 Derivation of Equations

Ascent :

The weight of the vehicle at burnout (W_{BO}) may be written as:

$$W_{BO} = W_{CS} + W_{TD} + W_{PD_a} + W_{RP_a} \qquad (1)$$

where:

W_{CS} = crew station weight

W_{TD} = propulsion system thrust dependent weights

W_{PD_a} = ascent propulsion system propellant dependent weights

W_{RP_a} = ascent unused reserve propellant

The vehicle burnout weight may also be written as:

$$W_{BO} = W_L / \; e^{\Delta V_a / I_{sp}\, g} \qquad (2)$$

where:

W_L = vehicle lunar launch weight

ΔV_a = lunar launch ideal velocity change

I_{sp} = propellant specific impulse

Equations (1) and (2) may be combined as follows:

$$W_L / e^{\Delta V_a / I_{sp}\, g} = W_{CS} + W_{TD} + W_{PD_a} + W_{RP_a} \qquad (3)$$

(See Paragraph 2.4 for complete symbol list)

The ascent reserve propellant is equal to a percentage of the ascent consumed propellant or:

$$W_{RP_a} = C_1 W_{P_a}$$

where: W_{P_a} = ascent consumed propellant

The ascent propellant dependent weights consist of the ascent pressurization system and propellant tanks and may be written as:

$$W_{PD_a} = C_2(W_{P_a} + C_1 W_{P_a}) = W_{P_a} (C_2 + C_1 C_2)$$

Note that C_2 is a function of propellant tank pressure (see Appendix 6C).

The ascent consumed propellant is determined by:

$$W_{P_a} = W_L - W_{BO} = W_L \left[1 - \frac{1}{e^{\Delta V_a / I_{sp} g}} \right]$$

Therefore, Equation (3) can be written as:

$$W_L / e^{\Delta V_a / I_{sp} g} = W_{CS} + W_{TD} + W_L (C_1 + C_2 + C_1 C_2) \left[1 - \frac{1}{e^{\Delta V_a / I_{sp} g}} \right]$$

or:

$$W_L \left[\frac{1 + C_1 + C_2 + C_1 C_2}{e^{\Delta V_a / I_{sp} g}} - (C_1 + C_2 + C_1 C_2) \right] = W_{CS} + W_{TD} \qquad (4)$$

The procedure used to Solve Equation (4) was to:

(a) Assume a ΔV_a
(b) Read the corresponding ascent thrust-to-weight ratio (T/W) from Figure 6-6.
(c) Assume W_L
(d) Calculate the required ascent thrust
(e) Read the required thrust dependent weights from the data presented in Appendix 6B
(f) Solve for W_L
(g) Iterate until the value of W_L assumed in Step (c) correspsonds to the value calculated in step (f).

This procedure gives the lunar launch weight for the assumed ΔV_a when solved in this manner.

Descent :

The weight of the vehicle at lunar landing is:

$$W_{LD} = W_L + W_{LG} + W_{PD_d} + W_{RP_d} + W_S - \frac{C_1}{2} W_{P_a} \qquad (5)$$

where: W_{LD} = vehicle lunar landing weight

W_{LG} = landing gear weight

W_{PD_d} = descent propellant dependent weight

W_{RP_d} = descent unused reserve propellant

W_S = weight of equipment staged on the moon (life support, secondary power, etc.)

W_{P_a} = ascent consumed propellant

The vehicle lunar landing weight may also be written as:

$$W_{LD} = W_o / \left(e^{\Delta V_h / I_{sp} g} \right) \left(e^{\Delta V_d / I_{sp} g} \right) \qquad (6)$$

where: W_o = LEM weight at initiation of gross deceleration

ΔV_h = ideal velocity change for hover, translation, and let-down

ΔV_d = ideal velocity change for gross deceleration

The weight of the landing gear may be determined as a percentage of the vehicle lunar landing weight as:

$$W_{LG} = C_3 W_{LD}$$

The descent reserve propellant is equal to a percentage of the descent consumed propellant or:

$$W_{RP_d} = C_1 W_{P_d}$$

where: W_{P_d} = descent consumed propellant

The descent propellant dependent weights consist of the descent pressurization system and propellant tanks and may be written as:

$$W_{PD_d} = C_2 (W_{P_d} + C_1 W_{P_d}) = W_{P_d} (C_2 + C_1 C_2)$$

The descent consumed propellant is determined by:

$$W_{P_d} = W_o - W_{LD} = W_o \left[1 - \frac{1}{(e^{\Delta V_h/I_{sp} g})(e^{\Delta V_d/I_{sp} g})} \right]$$

Therefore, Equation (5) can be written as:

$$\frac{W_o}{(e^{\Delta V_h/I_{sp}g})(e^{\Delta V_d/I_{sp}g})} = W_L + W_S - \frac{C_1}{2} W_{P_a} + W_o \left[\frac{C_3 \; -(C_1 + C_2 + C_1 C_2)}{(e^{\Delta V_h/I_{sp}g})(e^{\Delta V_d/I_{sp}g})} + (C_1 + C_2 + C_1 C_2) \right] \tag{7}$$

or:

$$W_o \left[\frac{1 + C_1 + C_2 + C_1 C_2 - C_3}{(e^{\Delta V_h/I_{sp}g})(e^{\Delta V_d/I_{sp}g})} - (C_1 + C_2 + C_1 C_2) \right] = W_L + W_S - \frac{C_1}{2} W_{P_a} \tag{8}$$

The procedure used to solve Equation (8) was to:

(a) Set lunar launch weight (W_L) to the value calculated from ascent considerations.

(b) Calculate ascent propellant (W_{P_a}) for the W_L from Step (a).

(c) Calculate $e^{\Delta V_h/I_{sp}g}$ since the ideal velocity change for hover and translation is independent of the gross deceleration trajectory.

(d) Assume a gross deceleration ideal velocity change (ΔV_d) from Figure 6-5.

(e) Solve for W_o.

This procedure gives the vehicle weight for the assumed ΔV_d with the following constraints:

(a) The descent propellant tanks and pressurization system are staged on the lunar surface.

(b) The thrust dependent weights are not staged on the lunar surface. It should be noted that the above equations assume that the thrust dependent weights calculated for ascent are adequate for descent. This assumption is seldom correct due to the different thrust levels required to accomplish the descent and ascent trajectories. This difference in thrust dependent weights requires an iteration between Equations (4) and (8) in order to determine the required thrust dependent weights.

(c) The reserve propellant philosophy is to provide the same percentage reserve (C_1) for both descent and ascent with the condition that one-half the ascent reserve will be available from unused descent reserve.

The difficulty in implementation of Equations (4) and (8) in determining the characteristics of a vehicle in regard to ascent and descent trajectories is a function of the type propulsion system utilized in the vehicle. A two-stage vehicle is quite simple to analyze since the ascent and descent trajectories can be treated completely independently. A one and one-half stage vehicle is quite complex since the thrust dependent weights must be matched for ascent and descent as noted in the above constraints. Solution of the equations becomes very complex if the propellant dependent weights are a function of the thrust-to-weight ratio, as is the case for a fixed area injector system. This complexity is due to the variation in required propellant tank pressure with throttle ratio (which is a function of descent thrust-to-weight ratio). A sample calculation for a one and one-half stage vehicle is presented in the following paragraphs.

2.3 Calculations

Conditions

(a) Storable propellant (I_{sp} = 320 sec.)
(b) One and one-half stage; i.e., descent propellant tanks and pressurization system staged on the lunar surface
(c) Three thrust chambers
(d) Maximum chamber pressure = 100 psia
(e) Fixed area injector with minimum pressure drop = 6 psi
(f) Nozzle expansion ratio = 40
(g) 10% reserve propellant for ascent and descent; i.e., C_1 = 0.10
(h) Ascent payload weight = 5187 lbs.
(i) Equipment staged on moon = 857 lbs. and C_3 = 0.05
(j) Hover, translation, and let-down ideal velocity change = 590 fps

Calculation of the characteristics of this type system is facilitated if a descent throttle ratio (hence a descent initial thrust-to-weight ratio) and a LEM weight at initiation of gross deceleration are assumed.

Ascent :

(a) Assume throttle ratio = 3.3:1 for each thrust chamber; therefore system throttle ratio = 9.9:1 and descent initial thrust-to-weight ratio = 0.65.

(b) The required propellant tank pressure is:

Tank pressure = chamber pressure + (injector pressure drop) x (throttle ratio)^2 = 100 + 6 $(3.3)^2$ = 165 psia

Therefore C_2 = 0.061.

Note: This value of C_2 was calculated for cylindrical tanks using the data presented in Appendix 6C.

(c) Assume W_o = 29,300 lbs.

(d) Calculate descent initial thrust level:

F_n = 0.65 (29,300) = 19,045 lbs.

(e) Read thrust dependent weights from Figure 6B-9.

W_{TD} = 739 lbs.

(f) Equation (4) can then be written as:

$$W_L \left[\frac{1 + 0.1671}{e^{\Delta V_a / I_{sp} g}} - 0.1671 \right] = 5187 + 739 = 5926$$

(g) Assume $\Delta V_a = 5805$ fps where:

$\Delta V = 5780$ for lunar boost

$\Delta V = \underline{25}$ for plane change

$\Delta V_a = 5805$ fps

(h) Calculate: $e^{\Delta V_a / I_{sp} g} = e^{5805/(320)(32.174)} = 1.7573$

(i) Equation (4) then becomes:

$$W_L \left[\frac{1.1671}{1.7573} - 0.1671 \right] = 5926$$

or:

$W_L = 11,921$ lbs.

Descent :

(a) With the lunar launch weight (W_L) calculated above, Equation (8) can be written as:

$$W_o \left[\frac{0.95 + 0.1671}{(e^{\Delta V_h / I_{sp} g})(e^{\Delta V_d / I_{sp} g})} - 0.1671 \right] = 11,921 + 857 - 0.05\, W_{P_a}$$

(b) The ascent consumed propellant (W_{P_a}) is:

$$W_{P_a} = W_L \left[1 - \frac{1}{e^{\Delta V_a / I_{sp} g}} \right] = 11,921 \; (1 - 0.5691) = 5137 \text{ lbs.}$$

(c) Since a descent initial thrust-to-weight ratio of 0.65 has been assumed, the ideal velocity change for gross deceleration is:

$\Delta V_d = 5875$ fps

Therefore, $e^{\Delta V_d / I_{sp} g} = e^{5875/(320(32.174)} = 1.7693$

(d) For a hover and translation ideal velocity change of 590 fps:

$$e^{\Delta V_h / I_{sp} g} = e^{590/(316)(32.174)} = 1.0597$$

Note: It was assumed that hover was accomplished with one engine in a throttled condition; hence,

$I_{sp} = 320(0.986) = 316$ sec. (see Figure 6B-14)

(e) Equation (8) can then be written as:

$$W_o\left[\frac{1.1171}{(1.0597)(1.7793)} - 0.1671\right] = 11,921 + 857 - 257$$

$$W_o = 29,307 \text{ lbs.}$$

The value of LEM weight at initiation of gross deceleration (W_o) calculated above checks the assumption of 29,300 lbs. in the ascent calculations; therefore, an iteration of the procedure is not necessary.

The above procedure was repeated for various values of ascent ideal velocity change to determine the minimum W_o for a descent throttle ratio of 3.3. The entire procedure was then repeated for other values of throttle ratio until the curves representing this system, as shown on Figure 6-8, could be constructed.

2.4 List of Symbols

C_1 - reserve propellant weight as a percentage of consumed propellant weight

C_2 - propellant dependent weight as a percentage of consumed propellant weight

C_3 - landing gear weight as a percentage of LEM lunar landed weight

F_n - propulsion system thrust level

g - gravitational constant - 32.174 ft/sec^2

I_{sp} - propellant specific impulse

ΔV_a - ideal velocity change for ascent

ΔV_d - ideal velocity change for gross deceleration

ΔV_h - ideal velocity change for hover, translation, and let-down

W_{BO} - LEM weight at ascent burnout

W_{CS} - crew station weight

W_L - LEM lunar launch weight

W_{LD} - LEM lunar landed weight

W_{LG} - landing gear weight

W_o - LEM orbiting weight prior to retro

W_{P_a} - ascent consumed propellant

W_{P_d} - descent consumed propellant

W_{PD_a} - ascent propellant dependent weights

W_{PD_d} - descent propellant dependent weights

W_{RP_a} - ascent unused reserve propellant

W_{RP_d} - descent unused reserve propellant

W_S - weight of equipment staged on moon (life support, secondary power, etc).

W_{TD} - propulsion system thrust dependent weights

3.0 SAMPLE CALCUATIONS - FINAL PROPULSION SYSTEM SELECTION

3.1 General

After the final propulsion system parametric studies were completed, vehicle design studies were initiated which utilized the three thrust chamber storable system described in Table 6-IV. These design studies indicated that the thrust chamber size created installation problems; therefore, the decision was made to reduce the thrust chamber size by increasing the chamber pressure to 150 psia and decreasing the nozzle expansion ratio to 30:1. The following system was then established for further study:

(a) Three ablative thrust chambers
(b) Storable propellants - N_2O_4, Aerozine-50
(c) Pressure feed
(d) Fixed injector
(e) Chamber pressure = 150 psia
(f) Nozzle expansion ratio = 30:1

This system was evaluated using the method described in Appendix 6E, Paragraph 2.0. The results of the evaluation, as shown in Figure 6-9, indicate that the LEM gross weight would be within Saturn C-5 payload capability. Concurrent design studies indicated that installation problems would not be encountered; hence, this system was recommended for the LEM.

3.2 Final System Design Point

The final system design point was a thrust-to-weight ratio of 0.65 at initiation of descent with a requirement for visual line-of-sight. Examination of Figure 6-9 indicates that the LEM gross weight at this design point would be approximately 29,900 lbs. The required throttle ratio is 3.3:1. Sample calculations for the final system are presented in the following paragraphs.

3.3 Calculations

Conditions :

(a) Descent ideal velocity change (ΔV):

Retro	255 fps	
Gross deceleration	5912	(See paragraph 4 for complete synbol list.)
Hover	320	
Translate and let-down	270	
Total =	6757 fps	

(b) Ascent ideal velocity change (ΔV_a):

Boost	5780 fps
Plane change	25
Total =	5805 fps

(c) Throttle ratio = 3.3:1
(d) Crew station weight (W_{CS}) = 4738 lbs.

(e) Thrust chamber pressure = 150 psia
(f) Nozzle expansion ratio = 30:1
(g) Nozzle cant angle = 11^{o}
(h) Storable propellants:
   Oxidizer - N_2O_4
   Fuel - Aerozine 50
   I_{sp} = 315 sec. nominal
   Due to the nozzle cant angle, a value of I_{sp} of 312 sec. was used for descent.

(i) One and one-half stage; i.e., descent propellant tanks and pressurization system stored on the lunar surface
(j) 10% reserve propellant for ascent and descent; assuming that 5% of the ascent reserve will be available from unused descent reserve.
(k) Ascent tanks and pressurization system will be capable of handling entire descent reserve propellant in the event it is unused.
(l) Fixed area injector with minimum pressure drop = 6 psia
(m) Equipment weight staged on moon (W_S) = 857 lbs.
(n) Landing gear weight (W_{LG}) = 975 lbs.

Calculations :

(a) Assume LEM orbiting weight = 29,860 lbs.

(b) The required thrust level (F_n) is:
   $F_n = 0.65\ (29,860) = 19,409$
   Therefore, thrust level is 6500 lbs. per chamber.

(c) The required propellant tank pressure is:
   Tank pressure = chamber pressure + (injector pressure drop) x (throttle ratio)^2 = $150 + (6)(3.3)^2$ = 215 psia
   Therefore, the sum of the propellant tank and pressurization system weights equal 7.1% of the propellant weight.

   Note: The value of 7.1% was assumed to be a nominal value for ascent and 0.4% was subtracted from descent calculations since the descent tanks are not insulated. Therefore, the following values were used:
   Descent - 6.7%
   Ascent - 7.1%

(d) The thrust dependent weights (W_{TD}) for F_n = 19,500 lbs. from Appendix 6B are:

Thrust chamber	360 lbs.
Lines, fittings, and valves	110
Gimbal and truss	25
Gimbal actuation	84
Total =	579 lbs.

(e) For the assumed LEM orbiting weight, the descent consumed propellant (W_{P_d}) is:

$$W_{P_d} = W_o \left[1 - \frac{1}{e^{\Delta V / I_{sp} g}}\right] = 29,900 \left[1 - \frac{1}{e^{6757/(312)(32.174)}}\right] = 14,630 \text{ lbs.}$$

Therefore, the descent reserve propellant is:
0.10 (14630) = 1463 lbs.

(f) The ascent propellant dependent weights (W_{PD_a}) are:

0.071 (1.05 x ascent consumed propellant + 1463)

(g) Assume the ascent consumed propellant (W_{P_a}) is 4842 lbs.

(h) The LEM ascent burnout weight (W_{BO}) is:

$$W_{BO} = W_{CS} + W_{TD} + W_{PD_a} + W_{PR_a}$$

(Reference Equation 1, Appendix 6E, Paragraph 2.0)

or:

$$W_{BO} = 4738 + 579 + 0.071 \left[1.05\ (4842) + 1463\right] + 484 = 6262 \text{ lbs.}$$

(i) The lunar launch weight (W_L) is:

$$W_L = W_{BO} \times e^{\Delta V_a / I_{sp} g} = 6262 \left[e^{5805/(315)(32.174)}\right]$$

$$W_L = 11,104 \text{ lbs.}$$

(j) Therefore, the ascent consumed propellant is:

$$W_{P_a} = W_L - W_{BO} = 11,104 - 6262 = 4842 \text{ lbs.}$$

Note that the calculated ascent consumed propellant agrees with the assumption in Step (g).

(k) The descent propellant dependent weights are:
0.067 (1.10 descent consumed propellant)

(l) Assume the descent consumed propellant weights 14,620 lbs.

(m) The lunar landed weight (W_{LD}) is:

$$W_{LD} = W_L + W_{LG} + W_{PD_d} + W_{RP_d} - 0.05\ W_{P_a} + W_S$$

(Reference Equation 5, Appendix 6E, Paragraph 2.0)

where: W_{RP_d} = descent reserve propellant

W_{P_a} = ascent consumed propellant

Therefore:

$$W_{LD} = 11104 + 975 + 0.067(16082) + 0.10(14620) - 0.05\ (4842) + 857$$

$$W_{LD} = 15{,}238 \text{ lbs.}$$

(n) Therefore, the LEM orbiting weight prior to retro (W_o) is:

$$W_o = W_{LD} \times e^{\Delta V / I_{sp} g} = 15238 \times e^{6757/(312)(32.174)}$$

$$W_o = 29{,}858 \text{ lbs.}$$

Note that this value agrees with the value assumed in Step (a).

(o) The descent consumed propellant (W_{P_d}) is:

$$W_{P_d} = W_o - W_{LD} = 29{,}858 - 15{,}238 = 14{,}620 \text{ lbs.}$$

Note that this value agrees with the values assumed in Step (e) and Step (1); therefore, the calculations are valid.

3.4 List of Symbols

C_1 - reserve propellant weight as a percentage of consumed propellant weight

C_2 - propellant dependent weight as a percentage of consumed propellant weight

F_n - propulsion system thrust level

g - gravitational constant = 32.174 ft./sec^2

I_{sp} - propellant specific impulse

ΔV - descent ideal velocity change

ΔV_a - ascent ideal velocity change

W_{BO} - LEM weight at ascent burnout

W_{CS} - crew station weight

W_L - LEM lunar launch weight

W_{LD} - LEM lunar landed weight

W_{LG} - landing gear weight

W_o	- LEM orbiting weight prior to retro
W_{P_a}	- ascent consumed propellant
W_{P_d}	- descent consumed propellant
W_{PD_a}	- ascent propellant dependent weights
W_{PD_d}	- descent propellant dependent weights
W_{RP_a}	- ascent unused reserve propellant
W_{RP_d}	- descent unused reserve propellant
W_S	- weight of equipment staged on moon (life support, secondary power, etc.)
W_{TD}	- propulsion system thrust dependent weights

ABSTRACT

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The results of propulsion system parametric studies for an Apollo Lunar Excursion Module are presented. Main propulsion and reaction control systems for use on this vehicle in the 1965 to 1966 time period are selected and described. A description of the operation of these two systems and a recommended development program for each system are included.